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RESEARCH MEMORANDUM

ANALYSIS OF THE TURBOJET ENGINE FOR PROPULSION OF SUPERSONIC FIGHTER AIRPLANES

By David S. Gabriel, Richard P. Krebs, E. Clinton Wilcox
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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS
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ANALYSIS OF THE TURBOJET ENGINE FOR PROPULSION OF SUPERSONIC FIGHTER AIRPLANES

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SUMMARY

An analysis has been made to determine the effects of various turbojet design and operating variables as well as various combinations of pertinent engine variables, or type of engine, on supersonic fighter or interceptor airplane performance. To cover the extremes of probable combat missions, two different types of flight plan were considered; one that involved primarily climb, acceleration, and supersonic combat, and a second that included 400 miles of cruise to combat and 30 minutes of hold or loiter time. A combat altitude of 50,000 feet was used and combat Mach numbers of 1.8 and 1.35 were considered, with principal emphasis on the Mach number 1.8.

The suitability or merit of the various engine designs and the relative importance of the different engine design variables are evaluated by the various airplane performance parameters. For engines providing adequate take-off and climb characteristics, and with fixed gross weight, pay load, and maneuverability requirements, the combat endurance affords a particularly critical figure of merit. Engine designs yielding greatest combat endurance are also capable of fulfilling a given combat mission with a minimum airplane gross weight.

The engine variables considered were compressor pressure ratio, compressor efficiency, turbine-inlet temperature, afterburner-outlet temperature, engine specific weight, air-handling capacity, type of exhaust nozzle, and type of engine installation (installed in nacelles or submerged in the fuselage). With each change in engine variable, appropriate changes in the airplane were made. The effects of individual variations in engine variables were studied; and, in addition, certain interrelated effects, such as the influence of variations of compressor variables on combustor velocities, are considered.

The results show generally that engine weight is an extremely important engine-design variable. Component efficiencies over the range generally encountered in present designs have somewhat less effect on

performance than engine weight. The use of a variable-area, convergent-divergent exhaust nozzle rather than a convergent nozzle, increases in turbine-inlet temperature, and increases in air-handling capacity produce large gains in performance.

Engines having component performance typical of currently available engines, that is, a turbine-inlet temperature of 2000° R, an air flow of 27 pounds per second per square foot of compressor-tip area, and a peak compressor efficiency of 85 percent, had an optimum afterburner temperature of 3500° R for both types of flight plan considered. Performance 2523
was generally not greatly affected by compressor pressure ratio. The insensitivity to compressor pressure ratio was unaltered by changes in turbine-inlet temperature, air-handling capacity, or type of exhaust nozzle.

For an engine with a turbine-inlet temperature of 2500° R or higher, an air-handling capacity of 33 pounds or more per second per square foot, and a variable-area, convergent-divergent exhaust nozzle, an afterburner may not be required.

Approximately the same type of engine provided maximum combat endurance, or minimum airplane gross weight, whether installed in nacelles or submerged in the fuselage. Similarly, variations in combat Mach number from 1.35 to 1.8 had only a small effect on the type of engine providing best combat endurance. If engines with advanced components are used, a combat endurance of 5 minutes may be realized for the short-range, or local defense, mission without appreciable cruise or loiter with an initial airplane gross weight of 11,000 pounds; and for the longer-range mission, with 400 miles of cruise radius plus hold or loiter time, with an initial gross weight of 20,000 pounds.

INTRODUCTION

With the advent of the turbojet engine, sufficient power per unit engine size became available to permit sustained supersonic flight. Concurrent advancements in airframe design have made supersonic flight a reality. For the potentialities of the turbojet engine to be fully realized for supersonic propulsion, both the over-all engine design and the operating conditions of its components must be properly selected for the particular flight conditions and propulsion requirements.

The high degree of interdependence of engine and airframe characteristics at supersonic flight speeds precludes adequate determination of engine design factors without consideration of the aerodynamic characteristics of the airplane in which it is installed. An analysis of aircraft propulsion systems must therefore be properly and completely integrated both with the characteristics of the airframe and with the particular propulsion requirements of the flight plan. One generalized

analysis of this type covering a wide range of flight and engine conditions is given in reference 1. The highly generalized nature of this work, however, while enabling a wide range of conditions to be covered, prevents direct application of the results to specific airplane types and the treatment of certain interrelated component operating limits.

An analysis has accordingly been conducted at the NACA Lewis laboratory to determine the effects of various engine design and operating parameters as well as various combinations of pertinent engine variables on the performance of supersonic aircraft having various specific flight plans and tactical missions. These airplanes include fighter or interceptor types having several different flight plans as well as long-range bombers. This report presents the results of the portion of the analysis dealing with the interceptor airplanes.

The most appropriate mission for these airplanes from tactical, strategic, and economic considerations is a matter of controversy and doubtless will vary as time and the state of the art progress. For the present investigation, therefore, two different missions covering a range of cruise radii and climb and maneuverability capabilities have been considered. These flight plans cover the extremes of probable combat missions and the results of the analyses will therefore bracket or include the requirements of intermediate flight plans. A combat altitude of 50,000 feet was used and combat Mach numbers of 1.8 and 1.35 were considered, with principal emphasis on the Mach number 1.8.

Representative airplane configurations were selected and the effect on the airplane performance of individual variations of the principal engine-design variables was computed. The suitability or merit of the various engine designs and the relative importance of the different engine design variables is judged by the various airplane performance parameters, such as take-off distance, time to combat, and combat endurance. For engines providing adequate take-off and climb characteristics and with fixed pay load and maneuverability requirements, the combat endurance affords a particularly critical figure of merit; conversely, engine designs that provide a maximum combat endurance for a given airplane gross weight are also capable of fulfilling a specified combat mission with an airplane of minimum gross weight.

The engine-design variables considered were compressor pressure ratio, compressor efficiency, turbine-inlet temperature, afterburner-outlet temperature, engine specific weight, and engine-air-handling capacity. Engines equipped with convergent and convergent-divergent exhaust nozzles were investigated. With each change in engine design variable, appropriate changes in the airframe were made, although with few exceptions the general configuration remained the same. These exceptions included investigations into the effects of nacelle or submerged installations.

ANALYSIS

Two basic flight plans, differing principally in the amount of subsonic flight, were considered. The first of these flight plans did not include subsonic cruise, loiter, or hold provisions, but was confined primarily to take-off, climb, acceleration, and supersonic combat. This mission is considered to be the extreme example of defense of localized areas from closely associated bases. The second mission included subsonic cruise or loiter together with appropriate hold and reserve provisions and is considered representative of the flight plan of a fighter or interceptor having the range capabilities necessary to defend 2523 the perimeter of a large area from widely separated bases.

For the first type of mission, combat Mach numbers of 1.8 and 1.35 were considered. The flight plans for the two combat Mach numbers were similar and are illustrated in figure 1. The flight was assumed to occur in the following sequence:

(a) The airplane made a ground run until its speed was 1.2 times the stall speed, at which speed take-off was accomplished. The airplane continued to accelerate at sea level until it reached a Mach number of 0.8.

(b) At a constant Mach number of 0.8 the airplane climbed to 35,000 feet.

(c) A second period of acceleration at constant altitude of 35,000 feet brought the fighter to design flight speed.

(d) The climb to combat condition, altitude of 50,000 feet, was made at the design Mach number.

(e) Consistent with the probable requirement of high rate of climb and maneuverability for a local defense interceptor, sufficient power was installed in the airplane to permit a 2g turn at the combat condition without loss of speed or altitude. When the combat altitude had been reached the plane went into a 2g maneuver. The turn radius at 50,000 feet in a 2g maneuver at a constant flight Mach number of 1.8 is 10.4 miles and at a constant flight Mach number of 1.35 is 5.8 miles. The engine thrust was held constant so that these radii decreased as the gross weight of the airplane decreased because of the reduction in gross weight due to the consumption of fuel. The airplane was assumed to combat until its fuel supply was exhausted. Afterburning was used throughout the flight.

The flight plan for the second type of mission is shown in figure 2, wherein altitude is shown as a function of distance. Take-off, acceleration, and climb to 35,000 feet were the same as for the first

interceptor mission. At 35,000 feet and a flight Mach number of 0.9 the airplane started a Breguet cruise which covered 400 nautical miles. At the end of cruise, acceleration to the combat Mach number of 1.8 occurred followed by climb to the combat altitude of 50,000 feet. In view of the subsonic cruise provisions and the reduced premium on very high rate of climb for this type of mission, the power loading was reduced from that of the local defense airplane. Accordingly, this airplane was powered to permit about a 1.5g turn at the combat condition without loss of
2523 speed or altitude. The airplane was assumed to combat in a 1.5g turn at a flight Mach number of 1.8 and an altitude of 50,000 feet. At the conclusion of combat, return to the base occurred at subsonic speed. Hold times of 15 minutes at 35,000 feet and a flight Mach number of 0.8 and 15 minutes at sea level at a flight Mach number of 0.3 were included. Afterburning was used in climb, acceleration, and combat, but all cruise and hold flight was with the afterburner inoperative.

Airplane Configurations

For all calculations except those in which the effect of gross weight was investigated, the airplane take-off gross weight was assumed to be 40,000 pounds and the pay load (pilot, electronic equipment, and armament) was 3000 pounds. In an analysis of this type it is possible to fix initial gross weight and compute endurance or combat time, or to fix combat time and compute initial gross weight. Both methods have some advantages. As will be shown, however, the effects of variations in the engine parameters and consequently the resulting engine designs for greatest combat endurance or minimum airplane gross weight are nearly the same. Because the assumption of constant initial gross weight is used principally in this report, the resulting combat times are in some cases considerably greater than required. For these cases initial gross weight could be reduced (not in direct proportion, however) until the combat time was decreased to the required value. In this manner combat time becomes a measure of the size or initial gross weight of the airplane as well as a measure of its endurance.

Representative airplane configurations were chosen for each combat Mach number. For the 1.8 Mach number airplane a straight tapered wing with modified hexagonal plan form and an aspect ratio of 3.0 was used. The wing thickness-chord ratio was 0.045. The drag of the tail was assumed to be 20 percent of the zero lift wing drag. A structure to gross weight ratio of 0.3 was assumed, consistent with contemporary fighter design. The structural weight was defined as gross weight minus engine weight, pay load, and fuel and fuel tank weights. The fuel tank weight was taken to be 0.1 of the fuel weight.

A tapered wing swept back 60° at the midchord and with a double-wedge section was used on the interceptor designed to combat at a Mach number of 1.35. The wing had an aspect ratio of 3.5 and a thickness-

chord ratio of 0.05. The structural to gross weight ratio was increased to 0.35 to account for the additional weight of the swept wing. The tail surface drag and fuel-tank weight assumptions were the same as for the 1.8 Mach number airplanes.

Two types of engine installation were considered, nacelle and submerged. For the nacelle installation it was assumed that two engines were used; and for the submerged installation, one engine was used. For the nacelle installations the fuselage was assumed to be a low-drag body with a length-diameter ratio of 12 and a frontal area of 13.3 square feet. The frontal area was chosen as approximately the minimum required for the pilot, armament, and radar equipment. The fuselage in all configurations was assumed to house most of the fuel.

The engine nacelles had a length-diameter ratio of 9, and the frontal area of the nacelles was 1.3 times the frontal area of the engine. The area swept by the compressor tip and the nacelle area were in the proportion 1:1.8, and the areas of both the primary combustors and the afterburner were enlarged to make the area of a circle enclosing these combustors 1.385 times the area swept out by the compressor tip.

For those configurations in which the power plant was submerged in the fuselage, the fuselage had to be enlarged to accommodate the engine as well as pilot, electronic equipment, armament, and most of the fuel. The method of selection of fuselage frontal area is discussed in appendix B. (The symbols used in appendix B and elsewhere are defined in appendix A.) This fuselage frontal area, for the size of engines encountered, afforded the necessary space for engine and pay load with sufficient clearance around the engine for mounting and cooling. The afterburner area was increased to 1.5 times the compressor tip area.

The wing loadings of the airplanes for the two different interceptor missions were set in different manners. For the first type of mission, that of a local defense interceptor, the wing loading was selected by the procedure described in appendix D so that lift-drag ratios in level flight and in a 2g maneuver were equal at the beginning of combat. This practice ensured adequate aerodynamic efficiency in both level flight and maneuvers and resulted in wing loadings ranging around 120 pounds per square foot. For the second, or area defense type of fighter, the optimum wing loadings for subsonic cruise and supersonic combat are considerably different. A study of the effects of wing loading was therefore made and as a result of this study a fixed wing loading of 100 pounds per square foot was found to be a good compromise value and was used exclusively for this airplane.

Range of Engine Variables

The thrust of the engines was fixed by the drag of the airplane during the required combat maneuver at the combat condition. The

size, weight, and fuel consumption of the engines were dependent, however, upon the engine design variables. The effects on airplane performance of independently varying the following engine design variables were computed:

Compressor pressure ratio at sea level, zero ram	3 to 15
Maximum compressor adiabatic efficiency	0.85 and 0.75
Air flow per unit of compressor frontal area (area swept by compressor tip), (lb/sec)/sq ft	18 to 36
[a]Turbine-inlet temperature, °R	2000 to 3000
Afterburner-outlet temperature, °R	turbine-outlet temperature to 4000
Engine specific weight, lb/sq ft compressor frontal area	400 to 800
Exhaust-nozzle velocity coefficient	1.0 to 0.9

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The following variables were held at the given constant values:

Primary-combustor efficiency	0.95
Ratio of primary-combustor total pressure loss to inlet total pressure	0.05
Ratio of primary-combustor diameter to afterburner diameter	1.00
Ratio of inner to outer diameter of primary combustor	0.40
Turbine efficiency	0.85
Ratio of afterburner friction pressure drop to burner inlet dynamic head	2
Afterburner combustion efficiency	0.85

For most of the calculations, use of a variable-area convergent exhaust nozzle was assumed. The effects on performance of using convergent-divergent nozzles with either a variable-area throat and a fixed-outlet area or with completely variable throat and outlet geometry were also determined. The performance with convergent-divergent exhaust nozzles was investigated assuming ideal one-dimensional flow in the nozzle and also under actual experimentally determined flow conditions.

The engine inlet diffuser was assumed to be of the spike type with a translating spike. The spike position was varied to minimize additive drag in the manner discussed in appendix C. The subsonic pressure recovery was assumed to be 0.95 of the total pressure after supersonic diffusion.

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[a]For turbine-inlet temperatures of 2500° and 3000° R, respectively, 2 and 5 percent of the engine air flow were assumed to be bled from the compressor discharge for turbine cooling.

It was assumed that the engines operated at maximum indicated engine speed and turbine-inlet temperature and with a constant afterburner-outlet temperature throughout the climb, acceleration, and combat portion of the flight plan. As the flight Mach number and altitude varied along the flight path, the compressor-inlet temperature and hence corrected engine speed varied. The variations in compressor efficiency and air flow with corrected engine speed given in reference 2 and in appendix C of this report were used to calculate engine performance during the flight. The compressor pressure ratio also varied with corrected engine speed or flight condition and was computed from the air flow and turbine-inlet temperature with the assumption that the turbine nozzles were choked. Details of the assumed component behavior are given in appendix C.

From this analysis it was possible to determine the independent effect on airplane performance of each of the engine design variables.

RESULTS AND DISCUSSION

A detailed discussion of the effect of the independent variation of engine design parameters on the performance of both the local and area defense interceptors designed to combat at a flight Mach number of 1.8 is first presented. Any difference in performance trends with engine parameters between the 1.8 and 1.35 Mach number design local defense airplanes is then shown. The principal discussion centers around the airplanes with power plants mounted in nacelles. Small changes in engine design brought about by submerging the engine in the fuselage are also discussed. Finally, the effect of simultaneous changes or improvements in engine design variables, that might be anticipated for future engines, is discussed.

Performance of Airplanes Designed for Mach Number of 1.8

Effect of afterburner temperature. - The first power-plant parameter investigated was the afterburner temperature. Increasing afterburner temperature with other engine design variables held constant increases engine thrust per unit frontal area and at the same time increases specific fuel consumption. The higher the afterburner temperature the smaller and lighter the engine necessary to supply the thrust for the airplane combat conditions. A double advantage is thus gained as afterburner temperature is increased. The engine becomes lighter, which permits carrying more fuel; and the engine becomes smaller, which reduces drag. The increased fuel consumption, however, tends to counteract these advantages.

The effect on the performance of the local defense interceptor of changing the afterburner temperature, and hence engine size, is shown

quantitatively in figure 3 where three airplane performance parameters, take-off distance, time required to reach the design point, and combat time, are plotted against afterburner temperature. The other power-plant parameters were fixed at the following values:

Compressor pressure ratio at sea level, zero ram	5
Maximum compressor efficiency	0.85
Turbine-inlet temperature, °R	2000
Engine air-handling capacity, lb/(sec)(sq ft compressor frontal area)	27
Exhaust nozzle	variable-area convergent without losses
Engine specific weight, lb/sq ft (corresponding to a specific weight at sea level, zero ram of 0.448 lb/lb thrust without afterburning)	650

In all succeeding discussion, the compressor pressure ratio is given as the value at sea-level, zero-ram conditions, and the compressor efficiency is given as the maximum value.

The take-off distance and time required to reach design point both increase as the afterburner temperature is raised, as shown in figure 3. The maximum take-off distance shown is 2250 feet and the maximum time required to reach design point is almost 4 minutes. Both these values are probably satisfactory for the type of interceptor mission under consideration, and therefore neither of these performance parameters limits the afterburner temperature within the range being considered.

The effect of changing the afterburner temperature on the take-off distance and the time required to reach design point, both quantities involving portions of the flight plan in which the flight Mach number is less than the design Mach number, can be explained by examining the effect of Mach number on the thrust augmentation ratio obtained from afterburning. For a given afterburner temperature the thrust augmentation ratio increases with increasing Mach number. Further, the rate of increase of thrust augmentation ratio with Mach number increases with elevated afterburner temperature. As a result, if two engines are capable of delivering the same thrust at the design Mach number, the one with the lower afterburner temperature will have the higher thrust in the low Mach number region of the flight plan and will give shorter take-off distance and require less time to reach the design point.

The combat time (fig. 3) increases continuously, but at a decreasing rate, as the afterburner temperature is raised above a minimum value of 2400° R. With an afterburner temperature of 3500° R, about 11.1 minutes of combat time are afforded. Increasing the afterburner temperature an additional 500°, or to a value of 4000° R, increases the combat time only 1.5 minutes.

The combat time, as previously mentioned, is intimately connected with the gross weight of the airplane. The airplane under study had a gross weight of 40,000 pounds and a combat time of 11.1 minutes. If a combat time of only 5 minutes were desired, the gross weight could be reduced to about 23,000 pounds.

The effect of afterburner temperature on the performance of the interceptor when the cruise and loiter provisions are included in the flight plan is shown in figure 4. For these calculations the same values of engine design variables that were used for the local defense airplane in figure 3 were specified except that the rated compressor pressure ratio was 7. The trends of take-off distance, time to design, and combat time with afterburner temperature are similar to those for the local defense interceptor. Both take-off distance and time to design are, however, greater for the area defense airplane. Take-off distances up to 3200 feet and times to design as high as 7.5 minutes are obtained. For this airplane, the time to design is defined as the time to reach design altitude and flight speed if the airplane follows the local defense flight plan. The increases in these values over those for the local defense airplanes are largely due to the reduced power loading (discussed in the ANALYSIS section) of the area defense airplane. A take-off distance of the order of 3000 feet is probably not excessive and time to design of 7 to 7.5 minutes, although probably not acceptable for an airplane involving the defense of a highly localized region, is reasonable for an area defense airplane that involves a greater radius of action. The combat time, shown in the lower curve of figure 4, reaches a maximum value at an afterburner temperature between 3800° and 4000° R. A maximum combat time of 6.5 minutes, or about one-half that for the local defense interceptor, may be obtained for the area defense airplane. 2523

In general, a comparison of figures 3 and 4 shows that even though the magnitudes of the airplane performance parameters differ the trends of performance are similar for the two types of airplane and that the afterburner temperatures for maximum combat time are about the same.

Because the increase in combat time with an increase in afterburner temperature from 3500° to 4000° R is small, and because the afterburner cooling problem is greatly intensified as the gas temperature is increased over the aforementioned range, much of the remaining analysis has been computed for an afterburner temperature of 3500° R. This temperature is considered to be a good compromise between the optimum shown by the analysis and practical considerations.

Effect of compressor pressure ratio and compressor efficiency (local defense airplane). - The effect of compressor pressure ratio and compressor efficiency on the local defense airplane performance parameters is shown in figure 5. The variation of compressor efficiency and corrected

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air flow with engine speed is assumed, in this analysis, to be independent of rated compressor pressure ratio (see figs. 39 and 40). With a rated pressure ratio of 5, a maximum compressor efficiency of 0.85, and an afterburner temperature of 3500° R, the take-off distance is 2000 feet, the time required to reach design point is 3.6 minutes, and the combat time is 11.1 minutes. A decrease in compressor efficiency decreases the performance of the airplane, with a reduction of 10 points in maximum compressor efficiency reducing the combat time by about 10 percent. These same trends are also applicable to changes in turbine efficiency.

All three airplane performance parameters are relatively unaffected by changes in pressure ratio from 5 to 9. As pressure ratios decrease below 5, the take-off distance and time to reach design increase while the combat time decreases only slightly. On the basis of airplane performance alone there is little to choose between compressor pressure ratios of 5 and 9 for a given specific engine weight.

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<u>Effect of engine specific weight (local defense airplane)</u>. - Engine weight is extremely important in a high-performance interceptor. This fact is illustrated in figure 6(a) in which combat time for three different engine weights per unit of compressor area is plotted against compressor pressure ratio for a compressor efficiency of 0.85. In an airplane in which the gross weight is fixed, additional engine weight must displace fuel and reduce the combat time. An increase in engine weight of 33 percent, from 600 to 800 pounds per square foot of compressor area, reduces the combat time from 12.2 to 8 minutes, or approximately 35 percent. This change in combat time is about three times the change in combat time occurring when the pressure ratio is varied from 3 to 9 at a fixed engine weight.

In the event that engine specific weight varied with compressor pressure ratio, this factor would have to be considered in evaluating the effect of compressor pressure ratio on airplane performance. Such an evaluation could be made by use of a figure similar to figure 6.

The relative importance of engine weight and compressor efficiency can be seen from figure 6(b). If, by a change in compressor design, the engine weight can be reduced 10 percent without reducing the compressor efficiency more than 0.10, a net gain in combat time can be realized. Figure 5 has shown that such a reduction in compressor efficiency does not increase either the take-off distance or the time to design to excessive or undesirable values.

<u>Primary combustor and afterburner velocity considerations</u>. - There are other factors besides airplane performance which should be considered in the choice of compressor pressure ratio. Burner-inlet velocities are of great importance if high combustion efficiencies are to be attained.

The primary-combustor-inlet velocity and the afterburner-inlet velocity at the design condition are shown in figure 7 as functions of the compressor pressure ratio for a combustor frontal area 1.38 times the compressor tip area. Both velocities decrease as the compressor pressure ratio is increased. If, for example, it is desired to limit the combustor-inlet velocity to a value of about 100 feet per second, which is representative of some present design practices, a compressor pressure ratio of about 6 is required for an engine with an air flow of 27 pounds per second per square foot of compressor area and the given area relations. 2523

Combustion efficiencies of 0.85 have been measured in afterburners with inlet velocities up to 500 feet per second (ref. 3). For the engine considered in figure 6, the afterburner-inlet velocity is less than 500 feet per second for compressor pressure ratios greater than 3.5. Reduction of the compressor efficiency from 0.85 to 0.75 increases the afterburner velocities approximately 60 feet per second, so that a compressor pressure ratio of approximately 4.5 is required to reduce the velocity below 500 feet per second.

Consideration of the velocity alone at the afterburner inlet may be misleading. The inlet pressure is also important, and the parameter PT/V in some instances has been shown to be pertinent (ref. 4). Both the inlet total pressure, expressed as the tail-pipe pressure ratio, and the parameter PT/V are plotted against compressor pressure ratio in figure 8. Both functions maximize at compressor pressure ratios between 6.5 and 7. Within this range of compressor pressure ratios the afterburner combustion efficiency should be a maximum, with other factors, such as burner length or fuel-air ratio, being constant.

The previous discussion has been based on engine and nacelle geometry in which the frontal area of the primary combustor and afterburner was assumed to be 38 percent larger than the frontal area swept by the compressor blade tips. In order to provide clearance for structure, accessories, and cooling passages, a ratio of nacelle to compressor frontal area of 1.8 was provided. Of course, the drag of the nacelle with these engine proportions is greater than it would be if the combustor diameters could be reduced and engine accessories relocated so that nacelle diameter could more closely approach compressor diameter. In order to utilize smaller diameter combustors without increasing inlet velocities, higher combustor-inlet pressures would be required. These higher pressures could be provided only by increasing compressor pressure ratio. The effect of reducing the ratio of combustion-chamber diameter to compressor diameter was investigated for both the nacelle and fuselage cases. It was found that for both cases, at compressor pressure ratios greater than 5, reducing the ratio of combustion-chamber diameter to compressor diameter to a value of 1.0 had a relatively slight effect on combat time. For compressor pressure ratios less than 5, rather large decreases in combat time occurred for the

case where combustor diameter was equal to compressor diameter, due to the occurrence of thermal choking and the necessity for operating at reduced afterburner temperatures.

Effect of compressor design and engine weight (area defense airplane). - As shown in figure 9 the trends of combat time with pressure ratio for the airplane with the cruise and loiter provisions in the flight plan are similar to those of the local defense airplane. For
2523 a constant engine weight per unit of frontal area, combat time is not sensitive to pressure ratio. Increasing engine weight from 600 to 800 pounds per square foot results in a decrease in combat time from 7.5 to 3 minutes or about 57 percent. Engine weight is therefore of great importance for both the area defense and local defense airplanes. As shown by the dashed line in figure 9, compressor efficiency also has a very large effect on the combat time. A decrease in compressor efficiency from 0.85 to 0.75 results in a decrease in maximum combat time from 12 to 3 minutes or 75 percent for an engine weight of 400 pounds per square foot. The very large effect of compressor efficiency on the area defense airplane as compared with the moderate effect on the local defense airplane is due to the greater proportion of subsonic flight and associated sensitivity to specific fuel consumption of the area defense fighter.

Effect of turbine-inlet temperature. - In a turbojet engine with a fixed afterburner temperature, the jet thrust of the engine is a function of the pressure ratio across the exhaust nozzle. If the turbine-inlet temperature is raised, the same compressor work requirements will result in a smaller pressure drop across the turbine and a consequent increase in the thrust output of the engine.

The effect of increased turbine-inlet temperature on airplane performance was evaluated. All the increased engine thrust calculated from simple thermodynamic analysis is not available, because some losses are associated with bleeding the compressor-discharge air required to cool the turbine. The analysis of reference 5 was used as a basis for the assumption that a 5-percent bleed was required for a turbine-inlet temperature of 3000° R, 2 percent for 2500° R, and no cooling air for 2000° R. After the air bled from the compressor had been used to cool the turbine it was returned to the tail pipe. The effect of this compressor air bleed on power-plant performance is included in the results presented.

The effect of turbine-inlet temperature is shown in figure 10 wherein the three airplane performance parameters are plotted against turbine-inlet temperature for an engine with a compressor pressure ratio of 5 and an afterburner temperature of 3500° R. If it were possible to increase the turbine-inlet temperature from 2000° to 3000° R, the take-off distance would be reduced from 2000 to 1600 feet. The same change in turbine-inlet temperature would reduce the time to design about

1/4 minute. Combat time (solid line) would be increased almost 18 percent from 11.1 to 13.1 minutes. Thus, it can be seen that increasing the turbine-inlet temperature affords a means of simultaneously improving all three of the airplane performance parameters. Most of the improvement can be realized by increasing the turbine-inlet temperature to 2500° R. For this temperature, the combat time, time to reach design, and take-off distance are 12.8 minutes, 3.3 minutes, and 1650 feet, respectively.

In figure 11, the effect of rated compressor pressure ratio on combat time of both the long- and short-range airplanes is shown for several engine specific weights and for a turbine-inlet temperature of 2500° R. An afterburner temperature of 3500° R is assumed for all conditions. The performance for the longer-range interceptor is shown in part (a) and for the shorter-range airplane, in part (b). The effect on combat time of increasing the turbine-inlet temperature from 2000° is evident from a comparison of figures 6 and 9 with figure 11. For a given engine specific weight, increasing the turbine-inlet temperature provides somewhat larger increases in combat time for the long range interceptor than for the shorter range airplane. Whereas the combat time, for a given engine specific weight, is virtually unaffected by changes in compressor pressure ratio from 5 to 15 for the short-range airplane, combat time for the longer-range airplane, at a constant engine specific weight of 600 pounds per square foot, is approximately doubled by increasing the compressor pressure ratio from 5 to 12.

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In figure 12, combat time is shown as a function of rated compressor pressure ratio for both the long- and short-range airplanes. An afterburner temperature of 3500° R was assumed, and curves are shown for turbine-inlet temperatures of 2000° and 2500° R. The engine specific weight for these calculations was independent of compressor pressure ratio or turbine-inlet temperature and was 650 pounds per square foot. Figure 12 illustrates the rather large gains made possible by increasing the turbine-inlet temperature. For a compressor pressure ratio of 12, increasing the turbine-inlet temperature from 2000° to 2500° R increases the combat time for the area defense airplane from 5.5 to 11.2 minutes. The corresponding increase for the local defense interceptor is from 10.4 to 14 minutes.

At the top of figure 12 are shown afterburner-inlet velocities for turbine-inlet temperatures of 2500° and 2000° R. The engines considered (having air-flow-handling capacity of 27 lb/sec/sq ft) have afterburner-inlet velocities of less than 500 feet per second for all compressor pressure ratios greater than 5. The primary-combustor-inlet velocities are, of course, unaffected by turbine-inlet temperature, except for the small decrease brought about by bleeding cooling air. Consequently, a compressor pressure ratio of 6 or higher should provide both satisfactory primary-combustor-inlet velocities and satisfactory afterburner velocities for a range of turbine-inlet temperatures up to 2500° R.

The effect of turbine-inlet temperature on airplane performance has been illustrated for an afterburner temperature of 3500° R. Of course, as turbine-inlet temperature increases, afterburner-inlet or turbine-outlet temperatures also increase. It might be expected, therefore, that the optimum afterburner temperature would decrease with increasing turbine-inlet temperature, and it is conceivable that with very high turbine-inlet temperatures the nonafterburning engine would equal the afterburning engine in performance. A study of the effect of afterburner temperature at turbine-inlet temperatures up to 3000° R showed, however, that performance without afterburning was inferior to that with afterburning, and that the best afterburner temperature remained about 3500° R if the engine were equipped with a convergent exhaust nozzle. As will be shown, the use of a convergent-divergent nozzle alters the results completely.

Effect of convergent-divergent nozzle. - A large pressure ratio across the exhaust nozzle is developed by turbojet engines operating at Mach numbers of 1.8. Exhaust-nozzle pressure ratios greater than 8 are shown in figure 8 for engines with compressor pressure ratios between 5 and 9. A convergent-divergent exhaust nozzle utilizes this pressure energy more effectively than does a simple convergent nozzle and thereby produces increased thrust. The enlarged exhaust nozzle outlet area also reduces the boattail drag, as indicated in appendix B.

Some of the effects on the performance of the short-range interceptor of using a continuously variable convergent-divergent nozzle without losses rather than a convergent nozzle are shown in figure 13, where the three airplane performance parameters are plotted against afterburner temperature. The engine design variables used in these calculations were compressor pressure ratio, 5; turbine-inlet temperature, 2000° R; air flow per unit compressor area, 27 pounds per second per square foot; maximum compressor efficiency, 0.85; and specific engine weight, 650 pounds per square foot. Because the engine thrust with a convergent-divergent nozzle is markedly improved over the thrust with a convergent nozzle at high Mach numbers, smaller engines may be used. Consequently, a deterioration in low-speed performance occurs. The take-off distance is increased by approximately 300 feet and the time to reach design point is increased about 1 minute by the use of a convergent-divergent nozzle. The combat time is increased about 6 minutes, or approximately 50 percent, for an afterburner temperature of 3500° R. Because the low-speed performance of the engine with a convergent nozzle is good, the penalties put on take-off distance and time to design by the use of a convergent-divergent nozzle may not be so important as the increase in combat time.

The thrust produced by an afterburning turbojet engine maximizes, and the specific fuel consumption minimizes, except for secondary effects resulting from differences in combustion efficiency between the

primary and afterburner combustors, at that compressor pressure ratio which provides maximum pressure ratio across the exhaust nozzle. The variation of thrust and specific fuel consumption with compressor pressure ratio is similar for an engine equipped with either a convergent-divergent nozzle or a convergent nozzle. The variation of combat time with compressor pressure ratio for an afterburning engine equipped with a convergent-divergent nozzle will be similar to that previously shown in figure 6.

Computations were also made of the performance of the local defense 2523
airplane when equipped with an engine using elevated turbine-inlet temperatures and a convergent-divergent nozzle but no afterburner. The specific engine weight was assumed to be reduced 16 percent by the elimination of the afterburner. A comparison of the performance with two nonafterburning engines and one afterburning engine is shown in the following table. All engines used a compressor pressure ratio of 5 and a compressor efficiency of 0.85.

Power plant		Airplane performance		
Turbine-inlet temperature (°R)	Afterburner temperature (°R)	Take-off distance (ft)	Time to design (min)	Combat time (min)
2000	3500	2300	4.55	17.2
2500	[a]2150	1390	2.98	11.4
3000	[a]2580	1460	3.40	17.9

[a]No afterburner used.

The foregoing table shows the 3000° R turbine-inlet temperature, nonafterburning engine to give superior performance as compared with the 2000° R, afterburning engine in all respects when a convergent-divergent nozzle is used. The low-speed performance of the 2500° R, nonafterburning engine is superior to that of either of the other two engines, but the combat endurance is less. The 2500° R engines are the largest of the three engines. It is of interest that the 3000° R engine is most economical of fuel in terms of fuel required per minute of combat. The three engines, in order of decreasing turbine-inlet temperature, burn 755, 862, and 881 pounds of fuel per minute of combat, respectively.

The results just presented were obtained with a continuously variable nozzle with no losses. The effect of nozzle losses, expressed in terms of velocity coefficient, is shown in figure 14. The engine used had a compressor pressure ratio of 5, a compressor efficiency of 0.85, a turbine-inlet temperature of 2000° R, and an afterburner temperature of 3500° R. A convergent-divergent nozzle with a velocity coefficient of 0.92 gives essentially the same combat time as a loss-free convergent nozzle.

Because a convergent-divergent exhaust nozzle with both throat and exit areas continuously variable may not be mechanically realizable, the local defense airplane performance was computed for a power plant with a nozzle having a variable throat area and a fixed exit area. The performance calculations were repeated for a series of different sized exit areas. For each of these exit areas there was a flight Mach number in the tropopause at which expansion of the exhaust gases to ambient pressure was complete. This flight Mach number has been used as the abscissa in figure 15. The nozzle throat was continuously varied to maintain constant rotational speed and turbine-inlet temperature within the engine.

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The flow from an engine equipped with a convergent-divergent nozzle designed to give complete expansion at a flight Mach number of 1.8 will be considerably overexpanded at low Mach numbers such as those encountered in take-off and climb. Severe thrust losses result if the flow expands according to one-dimensional flow theory, and the static pressure shocks up to ambient pressure. Airplane performance with an exhaust nozzle in which the flow follows one-dimensional theory is indicated by the curves marked "one-dimensional" on figure 15.

The aforementioned curves are of strictly academic value for in an actual convergent-divergent nozzle in which the pressure ratio across the nozzle and the ratio between the throat and exhaust area are such that overexpansion exists, the ambient pressure is propagated upstream through the boundary layer. The boundary layer is thus thickened, the expansion ratio is effectively reduced, and the Mach number at which shock occurs is also reduced. Convergent-divergent nozzle performance with pressure ratios less than the design value is given in reference 6. These data were used in computing the airplane performance for the curves marked "actual" in figure 15.

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On that same figure airplane performance with a continuously variable convergent-divergent nozzle is indicated by "X," and performance with a variable-area convergent nozzle is approximated at an abscissa value of 0.2.

An examination of figure 15 shows that the performance of the airplane without cruise or loiter provisions equipped with a fixed-exit convergent-divergent exhaust nozzle is nearly equal to the performance which would be obtained if a continuously variable convergent-divergent nozzle were used. The take-off distance with a fixed-exhaust-area, variable-throat, convergent-divergent nozzle designed for complete expansion at a Mach number of 1.8 is 2570 feet, 270 feet more than for a continuously variable convergent-divergent nozzle. The corresponding times to reach design conditions are 5.25 and 4.55 minutes. The combat time with the fixed-exit nozzle is 17.2 minutes, just 0.6 minute less than that available with a continuously variable nozzle. The difference in combat time comes from the fuel saved during climb and acceleration by use of a continuously variable convergent-divergent nozzle.

Because the time to reach the design point taken from figure 15 for a fixed-exit-area nozzle designed for complete expansion at a flight Mach number of 1.8 is considered marginal, some compromise in nozzle design may be in order. If the expansion ratio is reduced so that complete expansion occurs at a flight Mach number of 1.4, the airplane performance is modified, as shown in the following tabulation of performance with a fixed-exit-area, variable-throat, convergent-divergent nozzle:

Flight Mach number for complete expansion	Take-off distance (ft)	Time to design (min)	Combat time (min)
1.8	2570	5.25	16.2
1.4	2300	4.7	15.4

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This comparison shows that for a sacrifice of about 1 minute of combat time, 1/2 minute can be saved in reaching the design point.

The preceding discussion of the fixed-exhaust area, variable-throat, convergent-divergent nozzle has been limited to the performance of the local defense airplane. In the area defense airplane wherein about equal amounts of fuel are burned in subsonic cruise and in combat, the thrust losses due to overexpansion become more significant. The effects of adding divergence to the nozzle and thereby improving the combat performance and penalizing the cruise performance are so balanced that changing the nozzle from a variable-area convergent nozzle to a variable-throat, fixed-exhaust-area, convergent-divergent nozzle designed for complete expansion at a Mach number of 1.4 has almost no effect on combat time (fig. 16). Because adding divergence to the exhaust nozzle increases only the take-off distance and time to design, it is concluded that if nozzle fabrication difficulties permit only the throat of a convergent-divergent exhaust nozzle to be variable, it is better to use a convergent nozzle on the area defense airplane. Some improvement in combat time could be realized with a continuously variable convergent-divergent nozzle.

Effect of air-handling capacity. - The principal remaining power-plant design factor investigated is the air-handling capacity of the engine. The advantage of increasing air flow and thereby reducing power-plant weight and nacelle drag for a given thrust is obvious. Increased burner velocities and attendant pressure losses, however, also accompany the increased air flow. The presence of favorable and unfavorable effects on performance associated with a change in air-handling capacity suggests the necessity of compromise and warrants an investigation of the effect of air-handling capacity on airplane performance.

Use of a power plant with increased air-handling capacity results in decreased drag. The aerodynamically cleaner airplane has decreased

power requirements at high Mach numbers. This decreased drag is more effective in reducing power requirements at high flight speeds than at low, with the result that the low-speed performance of the cleaner airplane is penalized.

The result is illustrated in figure 17 wherein take-off distance, time to reach the design point, and combat time are plotted against rated air flow per unit of compressor area for three different engine compressor pressure ratios. The take-off distance shows a minimum for the two pressure ratios 5 and 7, while the time to reach the design point increases continuously with air flow for all pressure ratios. For a compressor pressure ratio of 7, the take-off distance increases from 1600 feet at an air flow of 27 pounds per second per square foot of compressor area to a value of 2200 feet at an air flow of 36 pounds per second per square foot. For this same compressor pressure ratio and over the same range of air flows, the time to reach design point increases from 3.4 minutes to 4.1 minutes.

These changes in low-speed performance probably do not jeopardize the low-speed performance sufficiently to render it marginal. The effect of air-handling capacity on the combat time, however, is very great. For a compressor efficiency of 0.85, increasing the air-handling capacity from 27 to 36 pounds per second per square foot of compressor area increases the combat time from 11.1 to 17.2 minutes. The combat time with an engine having an air flow of 22 pounds per second per square foot is one-third the combat time available with an engine which handles 36 pounds of air per second per square foot.

The data presented in figure 17 illustrate the fact that high air-handling capacity is beneficial only if compressor efficiency is not sacrificed. No performance gain can be realized if by increasing the air-handling capacity from 29 to 33 pounds per second per square foot of compressor area the compressor efficiency is dropped from 0.85 to 0.75.. With the low compressor efficiency, thermal choking in the afterburner occurs as the air-handling capacity is raised. With an afterburner temperature of 3500° R, thermal choking occurs at air flows between 30 and 32 pounds per second per square foot over a range of compressor pressure ratios from 3 to 7 with an efficiency of 0.75. Were the air flow to be increased to 36 pounds per second per square foot, the afterburner temperatures at which choking occurred would be reduced to between 2400° and 2700° R. With a compressor efficiency of 0.85, thermal choking occurs below a compressor pressure ratio of 3, at an afterburner temperature of 3500° R, and an air flow of 32 pounds per second per square foot.

As shown in figure 18, afterburner- and primary-combustor-inlet velocities are proportional to air-handling capacity. If, for example, it is necessary to keep the afterburner velocity below 500 feet per

second with a compressor pressure ratio of 5, the maximum permissible air flow is 30.5 pounds per second per square foot. The resulting combat time is 14.4 minutes. Use of a compressor pressure ratio of 7 permits air flows as high as 33 pounds per second per square foot before an afterburner velocity of 500 feet per second is exceeded. The corresponding combat time has increased to 16.8 minutes. Primary-combustor-inlet velocities of about 115 feet per second result.

The foregoing discussion has centered around the parameter rated
air flow per unit of compressor area. The rather arbitrary choice of 2523
this parameter for describing air-handling capacity is permissible in a power plant and nacelle design in which the geometry is fixed. One of the causes for the decreasing slopes of the curves of combat time against air flow with increasing air-handling capacity in figure 17 is the pressure losses associated with high air flows in the afterburner. These losses and the afterburner velocity shown in figure 18 are direct functions of the afterburner area. Essentially, then, the primary concern in a consideration of the air-handling capacity is the air flow per unit of primary combustor and afterburner areas. In this fixed-geometry analysis, air flow per unit of compressor area can be converted directly to air flow per unit of afterburner area. An air flow capacity of 30 pounds per second per square foot of compressor area is equivalent to 21.7 pounds per second per square foot of afterburner area.

Effect of submerged or fuselage engine installation. - The combined friction and pressure drags of airplanes with engines in nacelles are greater than for airplanes with engines submerged in the fuselage. As a result, for a given volume of power plant, pay load, and fuel, the lift-drag ratios of the airplanes with fuselage engine installations are greater than for the nacelle type. In addition, the space available for the afterburner in the aft portion of the fuselage permits the use of an afterburner diameter which is larger (resulting in lower pressure losses) relative to the compressor diameter than in the nacelle case. In this analysis the afterburner flow area was assumed to be 8 percent greater for the fuselage installation than for the nacelle installation.

The principal effects, therefore, of submerging the engines in the fuselage on the optimum engine design and on the airplane performance are the effects of higher airplane lift-drag ratios and reduced afterburner pressure losses. Although an exact evaluation of the magnitude of increase in lift-drag ratio is difficult to make, it is believed that the methods of analysis given in appendixes B, C, and D yield sufficiently accurate results to demonstrate the significant trends.

The comparison between the local defense airplane performance with fuselage and nacelle engine installations is given in figure 19. Take-off distance, time to reach the design point, and combat time are plotted against the compressor pressure ratio for an engine having a turbine-inlet temperature of 2000° R and an afterburner temperature of

3500° R. The take-off distance and time to design are affected only slightly by the type of engine installation and the combat time is about 20 percent greater for the fuselage installation. The curves show, however, that the optimum compressor pressure ratio is unchanged. Figure 19 shows performance with a constant specific engine weight. The increase in afterburner flow area of 8 percent for the fuselage installation results in a corresponding decrease in afterburner-inlet velocities. At a compressor pressure ratio of 4 the afterburner-inlet velocity is less than 500 feet per second.

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The effect of afterburner temperature on the airplane performance for fuselage and nacelle installations is shown in figure 20. The trends are similar in both cases but the combat time optimizes at a slightly lower value of afterburner temperature for the fuselage installation. A similar result is shown in figure 21 in which the effects of air-handling capacity are compared for the two types of installation. In the fuselage installation the gains in performance for increases in air-handling capacity are somewhat less than those for the nacelle installation.

These results all indicate that, although the combat time capabilities of the airplane with a fuselage installed engine are superior to those of the airplane with a nacelle installation, the optimum engine design is little affected. In addition, consideration must also be given to the size of the engine involved. Inasmuch as two engines are used in the nacelle installation and only one in the fuselage installation, an engine of approximately twice the size of the nacelle engines is required for the fuselage installation for the same gross-weight airplanes. In order to execute a 2g maneuver at an altitude of 50,000 feet and a flight Mach number of 1.8, a 40,000-pound gross-weight airplane requires an unaugmented thrust of approximately 25,000 pounds at take-off.

Effect of Combat Mach Number

In order to illustrate the effect of design flight speed on optimum turbojet power-plant design, portions of the analysis previously described for the local defense airplane with a design flight Mach number of 1.8 were repeated for a design flight Mach number of 1.35. Decreasing the design flight Mach number resulted in a substantial decrease in power-plant size; the required sea-level, zero Mach number unaugmented thrust output decreased from 25,000 pounds for the 1.8 Mach number interceptor to 18,000 pounds for the interceptor designed for a flight Mach number of 1.35.

The relative performance of the short-range interceptor aircraft designed for flight Mach numbers of 1.35 and 1.8, respectively, can be

seen by comparison of figures 3 and 22. For aircraft powered by engines having an afterburner-outlet temperature of 3500° R, the two airplanes have approximately equal take-off distances. Because of the lower thrust requirements of the 1.35 Mach number design airplane, 5.5 minutes are required to reach the design condition as compared with 3.6 minutes for the 1.8 Mach number design airplane. For an engine specific weight of 600 pounds per square foot of compressor area, the combat time for the lower-speed airplane is 27.7 minutes as compared with 12.2 minutes for the higher-speed airplane. The increased combat time is due partly to the decreased fuel consumption at the lower design Mach number and 2523
partly to the increased fuel weight carried in the 40,000-pound airplane at the lower design Mach number. The fuel weight increases for the lower design speed because the engine size and weight are reduced, as previously discussed.

Effect of afterburner temperature. - The effect of afterburner temperature on take-off distance, time to reach design condition, and combat time is shown in figure 22. In general, the effects are similar to those presented in figure 3 for the 1.8 Mach number design airplane. For an engine specific weight of 600 pounds per square foot of compressor area, the combat time for the 1.35 Mach number design reaches a maximum at an afterburner temperature of about 3700° R, although combat time is relatively insensitive to afterburner temperature in the range from 3300° to 4000° R.

Effect of engine specific weight. - The effect of engine specific weight on combat time is also shown in figure 22, where combat time is shown as a function of afterburner temperature for engine specific weights of 400, 600, and 800 pounds per square foot of compressor area. Time to reach the design condition and take-off distance are unaffected by changes in engine specific weight, inasmuch as increases or decreases in engine weight merely increase or decrease the available fuel load without affecting the engine power requirements. For the basic engine configuration with an afterburner temperature of 3500° R, increasing the engine specific weight from 600 to 800 pounds per square foot of compressor area, or 33 percent, decreases the combat time from 27.7 to 21.3 minutes, a decrease of about 23 percent. For the 1.8 Mach number design, with the same engine configuration, a similar increase in engine specific weight produced a reduction in combat time of 33 percent. The somewhat reduced effect of engine specific weight at the lower flight Mach number is a result of the decreased engine size; because the engine weight is a smaller percentage of the gross weight, a given change in engine specific weight has less effect on combat time.

Effect of compressor pressure ratio and compressor efficiency. - The previously discussed effects of afterburner temperature and engine specific weight have been based on an engine having a rated compressor pressure ratio of 5 and a compressor efficiency of 0.85. In figure 23, the take-off distance, time to reach design condition, and combat time

are each shown as a function of rated engine compressor pressure ratio for compressor efficiencies of 0.85 and 0.75 and several engine specific weights. A turbine-inlet temperature of 2000° R, an afterburner temperature of 3500° R, and an air-handling capacity of 27 pounds per second per square foot of compressor frontal area have been assumed. The time to reach design and the take-off distance both decrease somewhat as the compressor pressure ratio is increased for a compressor efficiency of 0.85. For a compressor efficiency of 0.75, both the time to reach design and the take-off distance are increased over corresponding values for the 0.85 efficiency case for all rated compressor pressure ratios.

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Decreasing the maximum compressor efficiency from 0.85 to 0.75 reduces the combat time by 4 and 7 minutes at rated compressor pressure ratios of 3 and 9, respectively. For the airplane designed for a Mach number of 1.8 the effect of compressor efficiency is considerably less. A decrease in compressor efficiency from 0.85 to 0.75 reduces the combat time 1 and 2.5 minutes at rated compressor pressure ratios of 3 and 9, respectively. For a constant engine specific weight, the combat time is relatively insensitive to rated compressor pressure ratio in the range from 5 to 9.

Effect of other engine design variables for 1.35 design Mach number. - Portions of the analysis for the 1.8 Mach number design were also repeated for the 1.35 Mach number design to determine the effects of turbine-inlet temperature and air-handling capacity. Although these results are not shown herein the general trends for both design Mach numbers were similar. Appreciable gains in combat time may be realized for the 1.35 Mach number design airplane by increasing both air-handling capacity and turbine-inlet temperature. It is of interest to note that for the 1.35 combat Mach number, the engine with a turbine-inlet temperature of 3000° R and no afterburning has at least equivalent performance to the engine with optimum afterburning temperature, regardless of the type of exhaust nozzle used.

The optimum engines for the 1.35 Mach number interceptor without cruise or loiter provisions are, in general, similar to those for the 1.8 Mach number airplane. Although an analysis of the effect of combat Mach number on the desirable engine characteristics for the interceptor including cruise and loiter provisions in the flight plan has not been made, it would be expected that an equal degree of congruity would exist.

Effects of Simultaneous Variations in Design Variables

The results discussed in the foregoing sections have indicated the effects on airplane performance of independently varying the different engine design variables in an effort to indicate the most desirable engine for application to a supersonic interceptor-type airplane using

currently available components. In the succeeding sections, the effects on airplane performance of simultaneously varying several engine design variables will be discussed in an effort to indicate the most desirable design characteristics to be incorporated in future engines for supersonic interceptor application.

Effect of turbine-inlet temperature. - The effect of turbine-inlet temperature on the optimum afterburner temperature for both the short- and long-range interceptors with nacelle-mounted engines is shown in figure 24. The engines considered have continuously variable convergent- 2523
divergent exhaust nozzles, an air-handling capacity of 33 pounds per second per square foot, a peak compressor efficiency of 0.85, and a compressor pressure ratio of 7. Lines are shown for the combat time available with engines having afterburner-outlet temperatures of 3500° R and those having no afterburner. The elimination of the afterburner has been assumed to reduce the engine weight 16 percent. As shown by the upper lines, the afterburner is necessary to provide optimum combat time for the local defense airplanes for turbine-inlet temperatures less than 2750° R. With a turbine-inlet temperature of 2500° R and an afterburner temperature of 3500° R, the combat time is 25.9 minutes. Eliminating the afterburner and increasing the turbine-inlet temperature to 3000° R results in a combat time of 28.0 minutes.

For the longer-range airplanes the intersection of the afterburning and nonafterburning curves occurs at a lower turbine-inlet temperature. The combat time for this airplane without afterburning is greater than the combat time with afterburning for all turbine-inlet temperatures higher than 2400° R.

The results of figure 24 demonstrate that engines having advanced components may not require afterburners for any application within the range covered by the assumed flight plans if the turbine-inlet temperatures are 2500° R or higher; however, the inclusion of an afterburner may still be desirable as an augmentation method to allow short periods of increased performance. In order to define completely the desirable components of these advanced engines, an examination of the effects of compressor pressure ratio is required.

Effect of compressor pressure ratio. - Figures 25 and 26 illustrate the effects of compressor pressure ratio on the performance of the short- and long-range interceptors, respectively, for the advanced engines of figure 24. The turbine-inlet temperature is 2500° R and no afterburning is used. As discussed previously, the attainment of satisfactory velocities into the primary combustors of this high air flow engine would probably require compressor pressure ratios of at least 7 or 8. For an engine specific weight of 600 pounds per square foot, the longer-range interceptor (fig. 26) has a maximum combat time of about 20 minutes at a compressor pressure ratio of 12.

<u>Effect of gross weight</u>. - As shown in figures 25 and 26, combat times of over 20 minutes for the local defense airplane and over 17 minutes for the area defense airplane are possible for the assumed initial gross weight of 40,000 pounds if these advanced engines are used. These combat times are considerably greater than the combat time required for a normal fighter or interceptor mission. If combat time is reduced, airplane gross weight may be reduced, as shown in figure 27. Combat time is plotted against initial gross weight for both types of interceptor mission. The engines considered in these calculations are the advanced engines with air-handling capacity of 33, peak compressor efficiency of 0.85, continuously variable convergent-divergent exhaust nozzles, compressor pressure ratio of 7, and turbine-inlet temperature of 2500° R. Although an afterburner temperature of 3500° R was used, approximately the same results would be obtained if no afterburner were used. The gross weight of the local defense airplane may be reduced to about 11,000 pounds if combat time is reduced to 5 minutes, and the gross weight of the area defense airplane may be reduced to about 20,000 pounds for a reduction in combat time to 5 minutes. It is of interest that at these low gross weights, at which combat time is relatively sensitive to gross weight, a difference in combat time of 1 minute or about 20 percent results in a difference in gross weight of about 5 percent.

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In view of the large difference in gross weight between the originally assumed value of 40,000 pounds and the gross weights required for 5 minutes of combat time if advanced engines are used, some of the calculations of the effect of engine design variables on airplane performance have been repeated for a reduced gross weight in order to demonstrate the validity of the performance trends established for the higher gross weights. The results of these calculations for an initial gross weight of 13,000 pounds are shown in figures 28 and 29.

In figure 28 combat time for the short-range interceptor is plotted against afterburner temperature and lines are shown for initial gross weights of 40,000 and 13,000 pounds. The engine considered had an air-handling capacity of 33 pounds per second per square foot of compressor frontal area, a turbine-inlet temperature of 2500° R, a peak compressor efficiency of 0.85, a compressor pressure ratio of 7, and a continuously variable convergent-divergent exhaust nozzle. The slope of the curve for 13,000 pounds gross weight is greater than the slope of the curve for the higher gross weight. The difference in slope is due to the relatively greater effect of engine weight on performance of the lightweight airplane. For the lower gross weight, the engine weight is a greater proportion of the total weight; hence a given percentage change in engine weight has a proportionately larger effect on performance. For both gross weights, however, nearly optimum performance is obtained at an afterburner temperature of 3500° R. Thus the essential trends of airplane performance with afterburner temperature are not altered appreciably by changes in gross weight.

The performance of both the light and heavy airplanes with nonafterburning engines is shown by the circled points. The nonafterburning engine, because of its lower thrust output, is, of course, heavier than the engine with a 3500° R afterburner temperature. Engine weight effects are therefore accentuated for the nonafterburning engine and become of greater importance as gross weight is decreased. With the 2500° R turbine-inlet temperature assumed, the reduction in combat time if an afterburner is not used (as indicated by the point labeled A) is less than 15 percent for an initial gross weight of 40,000 pounds, as previously shown in figure 24. At the gross weight of 13,000 pounds, however, a reduction in combat time (point B) of over 60 percent must be taken if an afterburner is not used. The turbine-inlet temperature for equal combat time with and without afterburning for the local defense interceptor would probably not change appreciably, however, from the value given in figure 24. These effects, although not negligible, do not invalidate the general trends and results previously presented for a fixed 40,000-pound initial gross weight.

The effect of compressor pressure ratio on combat time for the short-range interceptor with initial gross weights of 40,000 and 13,000 pounds is shown in figure 29. In these curves engine weight is constant at a value of 650 pounds per square foot of compressor tip area, and the engine design variables are the same as those for figure 30. An afterburner temperature of 3500° R was assumed. In general, the trends are the same for both gross weights.

The results of figures 28 and 29 show that the general airplane performance trends with changes in engine design for a wide range of airplane gross weights are similar, although the effects of engine weight are accentuated.

SUMMARY OF RESULTS AND CONCLUSIONS

An analysis has been made to determine the effects of turbojet design and operating variables on the performance of high altitude supersonic interceptor or fighter airplanes. This analysis indicates that engine weight is one of the most important parameters in the selection of the interceptor power plant. Engine component efficiencies within the range of variation encountered in current engine designs are less important than engine weight. The other engine design variables, such as compressor pressure ratio, turbine-inlet temperature, afterburner temperature, and air-handling capacity, interact in a complicated manner, and individual optimum values of design variables may not be selected unless all other design parameters are specified. For example, any change in an engine parameter, such as turbine-inlet temperature or air-handling capacity, that results in a lighter engine for the same thrust output reduces the optimum afterburner temperature.

Due consideration must also be given to the effect of changes in engine design variables on the operating conditions of the components, such as the velocity and pressure at the inlet of the primary combustor and the afterburner.

Currently available components are, in general, limited to turbine-inlet temperatures of 2000° to 2200° R, air flows of 27 to 30 pounds per second per square foot of compressor frontal area, and a maximum compressor efficiency of 0.85. For engines having these component limitations, an afterburner temperature of 3500° R is most desirable for both the short- and long-range interception missions.

Advancements in engine design that would make realizable the use of a continuously variable convergent-divergent exhaust nozzle will result in a large increase in combat time at some expense of low-speed performance.

Increasing the turbine-inlet temperature affords a means of improving the performance of both the local defense and area defense airplanes in all phases investigated. The best afterburner temperature is 3500° R for both airplanes if a convergent nozzle is used.

If air-handling capacity is increased to 33 pounds per second per square foot of compressor tip area, a compressor pressure ratio of at least 7 is probably required to obtain satisfactory primary-combustor-inlet velocities.

The use of a high turbine-inlet temperature in connection with increased air-handling capacity and a convergent-divergent exhaust nozzle may provide optimum airplane performance without an afterburner. If the turbine-inlet temperature is increased to about 2500° R for the long-range interceptor and to about 2750° R for the short-range interceptor, the same combat time is obtained with or without afterburning. Further increases in turbine-inlet temperature make the performance without afterburning superior to the performance with afterburning.

For the assumption used in this analysis, that the off-design performance of the compressor and the engine weight are independent of rated compressor pressure ratio, the effect of compressor pressure ratio on airplane performance was secondary in comparison with the effects of engine specific weight and specific air flow.

The short-range interceptor with the advanced engines would have a combat time of about 5 minutes for an initial gross weight of 11,000 pounds. The longer-range interceptor with the same engines would have 5 minutes combat time for an initial gross weight of about 20,000 pounds.

The general trends and results of the analysis of the effect of engine design variables on airplane performance are substantially independent of airplane initial gross weight; the principal exception is that the effects of engine weight are somewhat accentuated for lower initial gross weights.

For the short-range interceptor, improvements in performance approaching that obtainable with a convergent-divergent exhaust nozzle having both variable throat and expansion ratio may be obtained with a fixed-exhaust-area, variable-throat nozzle. In the longer-range interceptor, 2523 however, the large losses in nozzle efficiency during subsonic flight if a fixed-area convergent-divergent exhaust nozzle is used penalize the performance to the degree that no benefit may be obtained by using a convergent-divergent exhaust nozzle unless it has both variable throat and expansion ratio.

Installation of the engines in either nacelles or fuselages results in approximately the same optimum engine designs. The values of engine design variables which provide maximum combat time for the 1.35 Mach number airplane are similar to those for the engine in the 1.8 Mach number airplane.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, September 16, 1953

APPENDIX A

SYMBOLS

The following symbols are used in this report:

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A	area
AR	aspect ratio
B	constant in expression for fuselage frontal area, 13.3 sq ft
C,C'	aerodynamic coefficients
D	drag
e	wing efficiency
F	net thrust
$K,K',K'',K''',$ K_1,K_2	constants
L	lift
M	Mach number
q	dynamic pressure
S	wetted area
W_a	air flow
W_g	airplane gross weight at beginning of combat

Subscripts:

a	additive
b	fuselage
bt	boattail
c	compressor

cp	cowl pressure
D	drag
f	friction
i	inlet
j	jet
L	lift
n	nacelle
r	maneuverability factor
w	wing
0	zero lift

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APPENDIX B

AERODYNAMIC ASSUMPTIONS

The following assumptions were made in regard to the geometry, drag, and lift of the component parts of the airplanes used in the analysis.

Fuselage for Airplane with Engines in Nacelles

2523 The fuselage for the airplane with the engines in the nacelles was assumed to be a low-drag body with a fineness ratio of 12 and a maximum frontal area of 13.3 square feet. The pressure drag coefficient was computed for a Haack body by means of equation (14) in reference 7. This drag coefficient was assumed valid at a flight Mach number of 1.5, and was assumed to vary inversely with the square root of the Mach number (ref. 8) for Mach numbers greater than approximately 1.1. The pressure drag was assumed to reach zero at a Mach number of 0.95. The assumed variation of pressure drag with Mach number is shown in figure 30. The friction drag coefficient was computed from equation (9a) in reference 9. The Reynolds number was computed for an average altitude of 35,000 feet and a length of 36 feet and varied with Mach number. The variation of friction drag coefficient based on wetted area is shown in figure 31. The total drag coefficient for the fuselage based on frontal area is shown in figure 32.

Nacelles

The nacelles were assumed, for ease in drag calculations, to consist of three sections with an over-all fineness ratio of 9. The forebody had a fineness ratio of 3. The center section was cylindrical in shape with a fineness ratio of 3. The aft section had a similar fineness ratio and was of uniform diameter except for a 7.03° boattail which reduced the jet area to the required size.

Cowl pressure drag. - The cowl pressure drag at any Mach number was assumed to vary linearly with the inlet area ratio; that is, the ratio between the cross-sectional area at the lip of the diffuser and the maximum frontal area. The cowl pressure drag coefficient, based on maximum frontal area, for a Mach number of 1.8 is shown in figure 33 as a function of inlet area ratio. The drag coefficient is zero at an area ratio of unity. Experimental data points for cowl pressure drag taken from references 10 to 12 are shown on figure 33. Because spillage at the inlet was always behind an oblique shock and because mass-flow ratios exceeded 0.85, the small variation of cowl pressure drag coefficient with mass-flow ratio was neglected. Variation of cowl pressure drag coefficient with Mach number is as shown in figure 30.

Boattail drag. - Experimental pressure coefficients on a boattail at a Mach number of 1.91 were taken from figure 9(b) of reference 13. Average values of the pressure coefficients were integrated to give the drag coefficient, based on frontal area (shown in fig. 34), as a function of the boattail area ratio. The variation of boattail pressure drag coefficient with Mach number is the same as that of cowl pressure drag coefficient with Mach number.

Friction drag. - The friction drag coefficient, based on wetted area, was the same as for the fuselage in the nacelle airplane (fig. 31). 2523
This coefficient was multiplied by the ratio of the wetted area to the frontal area to give the drag coefficient based on frontal area.

Fuselage for Airplane with Engine Submerged

The fuselage for the submerged-engine airplane was considerably larger than the fuselage required for the airplane with engines mounted in nacelles. Although the absolute magnitude of fuselage frontal area is a principal factor in determining airplane drag and hence the absolute level of airplane performance, it was reasoned that in this case, in which the trends or variations in airplane performance with engine size were of greatest importance, to establish a valid variation of fuselage frontal area with engine frontal area was a primary requirement. Because very little data on the drags of the component parts of fuselages with scoop-type inlets were available, a brief study of fuselages with nose-type inlets was undertaken. Layouts of several fuselage arrangements were made with the specification that the air velocity in the engine inlet ducts passing the pilot should be reasonable. It was found that the variation in fuselage frontal area with compressor frontal area could be closely approximated by the relation

$$A_b = 13.3 + A_c \tag{B1}$$

This relation is limited in application to the range of engine and airplane sizes covered herein. If engine size becomes very small or very large relative to the airplane size, a linear variation is no longer valid.

For ease in calculating the drag of the fuselage, it was assumed to be made up of three sections - an open-nose cowl with a fineness ratio of 3, a center section of constant diameter and fineness ratio of 6, and an aft section of constant diameter with a 7.03° boattail of length sufficient to reduce the exit area to its required value; the fineness ratio of the aft section was 3. Pressure and friction drags were computed from figures 31, 33, and 34 as for the nacelle.

Checks of the drag calculations with the drag coefficients of airplane fuselages available in the literature indicated a very close agreement.

Wing for 1.8 Mach Number Airplane

The 1.8 Mach number airplane had a straight, tapered wing of modified hexagonal section. It had an aspect ratio of 3, a taper ratio of 0.4, and a thickness-chord ratio of 0.045. Interference effects between wing and fuselage were considered by assuming the effective area of the wing to extend to the fuselage center line. In computing the pressure drag and friction drag of the wing and tail, the combined area was taken as 1.2 times the wing area. The drag coefficient of the wing at zero lift $C_{D,0,w}$ was taken from figure 5 of reference 14.

The total drag coefficient for the wing and tail was then computed from

$$C_{D,w} = 1.2\ C_{D,0,w} + C_{D,L} \tag{B2}$$

where

$$C_{D,L} = \frac{\sqrt{M^2 - 1/4}}{e\left(1 - \dfrac{1}{2AR\sqrt{M^2 - 1}}\right)} C_L^{\ 2} \tag{B3a}$$

in the supersonic region, and

$$C_{D,L} = \frac{C_L^{\ 2}}{\pi ARe} \tag{B3b}$$

in the subsonic region. The wing efficiency e was assumed to be 0.93 for $M > 1.1$, and 7 for $M < 0.95$. The assumed drag at zero lift and drag due to lift as functions of Mach number are shown in figures 35 and 36, respectively.

Wing for 1.35 Mach number Airplanes

A tapered wing, swept 60° at the midchord and with a 5 percent thick double-wedge section, was used on the 1.35 Mach number airplane. The wing had an aspect ratio of 3.5. The tail area was assumed to be 0.20 of the wing area. The pressure drag for wind and tail at zero lift was taken from figure 2 of reference 7. A constant friction coefficient of 0.003 was assumed for the range of Mach number considered. The drag due to lift was taken from figure 7 of reference 15 and figure 12 of reference 16. The variation in the drag due to lift with lift coefficient was approximated by the analytical expression

$$C_{D,L} = K'C_L^{\ 2} + K''C_L + K''' \tag{B4}$$

where K', K'', and K''' are functions of flight Mach number and wing plan form and were evaluated empirically.

APPENDIX C

POWER-PLANT ASSUMPTIONS

The power-plant performance at rated engine speed was computed in terms of thrust per unit air flow and specific fuel consumption over a range of flight Mach number from 0 to 2.0. Assumptions on component behavior were as follows. 2523

Diffuser

A spike diffuser was used on all engines. The inlet area was chosen to give a mass-flow ratio of unity at the design point. For Mach numbers and altitudes other than design the spike was adjusted to give minimum additive drag (ref. 17). Under these assumptions the additive drag coefficient based on inlet area as a function of flight Mach number was as shown in figure 37.

The supersonic pressure recovery for the diffuser was calculated assuming the oblique shock and the normal shock to stand at the lip of the diffuser. A subsonic pressure recovery of 0.95 was assumed. The total-pressure recovery for the diffuser is shown in figure 38. The pressure recovery was assumed to be the same for both nacelle and fuselage engine installations.

Compressor

Consideration is given to the off-design performance of the compressor. The pressure ratio quoted in the text or used on the figures is the design pressure ratio, or the pressure ratio at rated speed, standard sea-level conditions and zero flight Mach number. The compressor-outlet pressure was computed from the engine gas flow and turbine-inlet temperature with the assumption of a choked turbine nozzle. Engines having design pressure ratios from 3 to 15 were considered.

The air flow expressed in terms of rated air flow, that is, the air flow at rated speed, zero Mach number at sea level, is shown as a function of generalized engine speed in figure 39. Two maximum compressor efficiencies, 0.85 and 0.75, were used in the analysis. Changes in compressor efficiency with engine speed are shown in figure 40 expressed in terms of the ratio of compressor efficiency to maximum efficiency and the ratio of generalized engine speed to the generalized engine speed at which maximum efficiency occurred (ref. 2). Peak efficiency was assumed at 0.80 rated generalized engine speed.

Primary Combustor

Combustion efficiency was assumed constant at 0.95. The total-pressure loss across the combustor was 0.05. Heating value of the fuel was 18,700 Btu per pound. The fuel consumption was computed from the fuel-air ratio charts of reference 18.

Turbine

Because operation of the engine was always considered at rated engine speed, the turbine nozzle was assumed to be always choked. Turbine efficiency was constant at 0.85.

Afterburner

Both friction and momentum pressure losses were computed for the afterburner. The friction drop was equal to twice the dynamic pressure at the inlet to the afterburner. The momentum pressure drop was based on the inlet Mach number and the temperature ratio across the afterburner. Combustion efficiency in the afterburner was 0.85.

Exhaust Nozzle

Exhaust nozzles of both the convergent and convergent-divergent type were used. The minimum area was determined to provide rated engine speed and turbine-inlet temperature, irrespective of the amount of afterburning employed. The exit area of the convergent-divergent nozzle was corrected to provide complete expansion down to ambient pressure, except in those instances where a nozzle of fixed exit area was used. A range of velocity coefficient from 0.9 to 1.0 was employed in the analysis.

Fuel System

The fuel tank weight was assumed to be 10 percent of the weight of fuel on board at take-off.

APPENDIX D

MATCHING OF POWER PLANT AND AIRPLANE

The problem of matching power plant and airplane wing size arises from the interdependence of power plant and airplane. The thrust required is determined by the airplane drag, but this drag is dependent in part on the drag of the power-plant housing, which, of course, is a function of the power-plant size. It therefore is desirable to 2523 secure a set of analytical relations between power plant and airplane parameters which yields a solution to power-plant size and airplane drag. This procedure is outlined in this appendix.

Short-Range Interceptor with Power Plant in Nacelles

<u>1.8 Mach number design</u>. - Three fundamental relations were used as a starting point. The first of these related the airplane drag to the drag at zero lift and the drag due to lift.

$$C_D = C_{D,O} + KC_L^2 \tag{D1}$$

where K is a function of the wing plan and KC_L^2 is identical to $C_{D,L}$ in equation (B3). The second of these was the relation between drag coefficient, lift coefficient, and lift-drag ratio.

$$C_D = \frac{C_L}{L/D} \tag{D2}$$

The third expressed the assumption that the lift-drag ratio at the design Mach number and altitude are the same when the airplane is in level flight and in a 2g turn.

$$\left(\frac{L}{D}\right)_{r=1} = \left(\frac{L}{D}\right)_{r=2} \tag{D3}$$

Simultaneous solution of the three foregoing equations resulted in solutions for C_L in level flight and L/D in terms of $C_{D,O}$ and K only.

$$C_L = \sqrt{\frac{C_{D,O}}{2K}} \tag{D4}$$

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and

$$\frac{L}{D} = \frac{1}{3}\sqrt{\frac{2}{KC_{D,O}}} \tag{D5}$$

(The lift-drag ratio as determined by eq. (D5) was within 6 percent of the maximum lift-drag ratio.) Determination of $C_{D,O}$ would result in a complete solution for lift coefficient and lift-drag ratio.

The drag coefficient at zero lift was written as

$$C_{D,O} = 1.2\ C_{D,O,w} + \frac{A_b}{A_w} C_{D,b} + \frac{A_n}{A_w} C_{D,n} \tag{D6}$$

in which the fuselage-wing and nacelle-wing area ratios are unknown. But

$$\frac{A_n}{A_w} = \frac{F/A_w}{W_a/A_n \times F/W_a} \tag{D7}$$

and

$$A_w = \frac{W_g}{qC_L} \tag{D8}$$

Because the thrust required for a 2g turn must be twice that in level flight, the maximum thrust requirement for the engine was given by

$$\frac{F}{A_w} = \frac{2W_g}{(L/D)A_w} \tag{D9}$$

Substituting equations (D8), (D4), and (D5) into equation (D9) and the resulting equation into equation (D7) resulted in

$$\frac{A_n}{A_w} = \frac{3qC_{D,O}}{\left(\frac{W_a}{A_n}\right)\left(\frac{F}{W_a}\right)} \tag{D10}$$

where W_a/A_n and F/W_a are known parameters of the power plant under consideration.

From equations (D8) and (D4) the ratio of fuselage to wing areas was written as

$$\frac{A_b}{A_w} = \frac{qA_b}{W_g}\sqrt{\frac{C_{D,O}}{2K}} \tag{D11}$$

When equations (D10) and (D11) were substituted into equation (D6) the result was the equation in $C_{D,O}$ and known power-plant parameters which was solved for $C_{D,O}$: 2523

$$C_{D,O} = \left\{\frac{\dfrac{qA_bC_{D,b}}{\sqrt{2K}\,W_g} + \sqrt{\left(\dfrac{qA_bC_{D,b}}{\sqrt{2K}\,W_g}\right)^2 + 4.8\,C_{D,O,w}\left(1 - \dfrac{3qC_{D,n}}{\dfrac{W_a}{A_n}\dfrac{F}{W_a}}\right)}}{2\left(1 - \dfrac{3qC_{D,n}}{\dfrac{W_a}{A_n}\dfrac{F}{W_a}}\right)}\right\}^2 \tag{D12}$$

In this equation all terms are constant for a given flight condition except the parameter $\dfrac{C_{D,n}}{\dfrac{F}{W_a}\dfrac{W_a}{A_n}}$ and the gross weight. The solution to the equation is presented graphically in figure 41. The zero-lift drag is insensitive to small variations in gross weight.

With the determination of $C_{D,O}$ the complete airplane configuration was obtained from the foregoing equations, and enough was known to determine the airplane drag at any Mach number.

<u>1.35 Mach number design</u>. - A similar matching procedure was used for the airplane designed for a Mach number of 1.35 with a sweptback wing. The total drag coefficient for the airplane was modified in accordance with the expression for drag due to lift for the swept wing (eq. (B4))

$$C_D = C_{D,O} + K'C_L^2 + K''C_L + K''' \tag{D13}$$

Short-Range Interceptor with Power Plant Submerged in Fuselage

For the airplane with the engine submerged in the fueslage the matching procedure had to be modified to provide an analytical determination of the airplane drag coefficient. The three fundamental equations (D1), (D2), and (D3) are valid for the fuselage installation; and

the resulting equations for lift coefficient (D4) and lift-drag ratio (D5) are the same. With only the wing-tail combination and the fuselage contributing to the airplane drag, the drag coefficient at zero lift was written as

$$C_{D,O} = 1.2\ C_{D,O,w} + C_{D,b} \frac{A_b}{A_w} \tag{D14}$$

2523 The fuselage area was given by the expression

$$A_b = B + A_c \tag{D15}$$

where B was taken as 13.3 square feet.

Solution of the seven foregoing equations for $C_{D,O}$ resulted in an equation similar to equation (D12) with B substituted for A_b, A_n replaced by A_c, and $C_{D,n}$ replaced by $C_{D,b}$. For the fuselage containing the engine the drag coefficient is unknown and is a function of the air flow, which is also undetermined until the drag of the airplane can be computed. Hence, the method fails unless a trial-and-error solution is tolerated.

In the actual method used to determine engine size and to match the wing area to the fuselage, use was made of equation (D14) and an approximate analytical expression for $C_{D,b}$ was written in terms of known power-plant parameters. The fuselage drag coefficient was made up of three components - cowl pressure, boattail, and friction drag.

$$C_{D,b} = C_{D,cp} + C_{D,bt} + C_{D,f} \frac{S}{A_b} \tag{D16}$$

where

$$C_{D,cp} = C'_{D,cp} - C'_{D,cp} \frac{A_i}{A_b} \quad \text{(fig. 32)} \tag{D17}$$

$$C_{D,bt} = K_1 \left[1 - \left(\frac{A_j}{A_b} \right)^{\frac{3}{2}} \right] - K_2 \left(1 - \frac{A_j}{A_b} \right) \tag{D18}$$

and

$$\frac{S}{A_b} = 2\sqrt{9.25 + 18\left(\frac{A_i}{A_b}\right)^{\frac{1}{2}} + 8.5\,\frac{A_i}{A_b} + 0.25\left(\frac{A_i}{A_b}\right)^2} + 36 + \frac{1 - \frac{A_j}{A_b}}{\sin\theta} - 2\,\frac{\left[1 - \left(\frac{A_j}{A_b}\right)^{\frac{1}{2}}\right]}{\tan\theta} \tag{D19}$$

where θ is boattail angle. For the fuselage with the engine submerged therein, geometrically constructed as described in appendix B, the ratio 2523
of surface area to frontal area could be approximated by a linear relation involving the inlet and outlet area ratios with $\theta = 7^{\circ}$.

$$\frac{S}{A_b} = 40.56 + 3.88\,\frac{A_j}{A_b} + 6.12\,\frac{A_i}{A_b} \tag{D20}$$

The inlet and outlet area ratios, A_i/A_b and A_j/A_b, were written as functions of A_i/A_c, A_j/A_c, W_g, F/W_a, W_a/A_c, B, K, and $C_{D,O}$, all of which were known except $C_{D,O}$.

$$\frac{A_i}{A_b} = \frac{A_i}{A_c}\,\frac{A_c}{A_b} \qquad \frac{A_j}{A_b} = \frac{A_j}{A_c}\,\frac{A_c}{A_b} \tag{D21}$$

where

$$\frac{A_c}{A_b} = \frac{6W_g}{B\left(\frac{F}{W_a}\right)\frac{W_a}{A_c}\sqrt{\frac{2}{KC_{D,O}}} + 6W_g} \tag{D22}$$

With the help of the relations (D16), (D17), (D18), and (D20), the fuselage drag coefficient was written as

$$C_{D,b} = (6.12\,C_{D,f} - C'_{D,cp})\,\frac{A_i}{A_b} + (K_2 + 3.88\,C_{D,f})\,\frac{A_j}{A_b} - K_1\left(\frac{A_j}{A_b}\right)^{\frac{3}{2}} + (C'_{D,cp} + K_1 - K_2 + 40.56\,C_{D,f}) \tag{D23}$$

a single equation with two unknowns.

Another equation involving the two unknowns $C_{D,b}$ and $C_{D,O}$ was evolved from the combination of equations (D4), (D5), (D8), (D14), and (D15), and the equation for air flow

$$W_a = \frac{2W_g}{\frac{L}{D} \times \frac{F}{W_a}} \tag{D24}$$

Thus,

$$C_{D,b} = \frac{C_{D,0} - 1.2\ C_{D,0,w}}{\frac{Bq\sqrt{C_{D,0}}}{\sqrt{2K}\ W_g} + \frac{3q\ C_{D,0}}{\frac{F}{W_a}\frac{W_a}{A_c}}} \tag{D25}$$

The right-hand sides of equations (D23) and (D25) were set equal to each other and solved for A_i/A_c with the aid of equations (D21) and (D22). The area ratio A_i/A_c is a function of A_j/A_c, F/A_c, and $C_{D,0}$. For a given airplane and flight condition, W_g, $C_{D,f}$, $C_{D,cp}$, K, K_1, K_2, $C_{D,0,w}$, B, and q are all fixed. The ratio A_i/A_c was plotted against $C_{D,0}$, and cross plots yielded $C_{D,0}$ as a function of F/A_c and A_j/A_c for a selected A_i/A_c. With the assumed inlet configuration the flight Mach number and air-handling capacity W_a/A_c determine A_i/A_c. In figure 42, $C_{D,0}$ is plotted against F/A_c for several values of A_j/A_c. This figure is for a Mach number of 1.8, an altitude of 50,000 feet, and an air-handling capacity of 27 pounds per second per square foot. For any engine under consideration, F/A_c and A_j/A_c are known and the drag coefficient at zero lift can be determined from a figure similar to figure 42.

The determination of $C_{D,0}$ together with equations (D4), (D5), (D8), (D15), and (D24) makes possible the determination of the complete airplane configuration.

Long-Range Interceptor with Power Plant in Nacelles

For the longer-range interceptor the wing loading was held at a fixed value. This necessitated a modification in the analytical procedure for matching the airplane and power plant. The total drag coefficient of the airplane was written as the sum of the drag coefficients of the wing and tail, the body, and the nacelles, multiplied by appropriate area ratios. Thus

$$C_D = C_{D,w+t} + \frac{A_b}{A_w} C_{D,b} + \frac{A_n}{A_w} C_{D,n} \tag{D26}$$

A substitution for the drag coefficient of the wing and tail yields

$$1.2\ C_{D,0,w} + KC_L^{\,2} + \frac{A_b}{A_w} C_{D,b} = C_D - \frac{A_n}{A_w} C_{D,n} \qquad \text{(D27)}$$

The same equation can be written in terms of the airplane drag q and the wing area as follows:

$$1.2\ C_{D,0,w} + KC_L^{\,2} + \frac{A_b}{A_w} C_{D,b} = \frac{D\ A_n}{qA_wA_n} - C_{D,n} \frac{A_n}{A_w} \qquad \text{(D28)}$$

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Equation (D28) may be solved for A_n/A_w; and when the thrust is substituted for the drag, the following equation results:

$$\frac{A_n}{A_w} = \frac{1.2\ C_{D,0,w} + KC_L^{\,2} + \frac{A_b}{A_w} C_{D,b}}{\frac{F/\delta_0}{(W_a\sqrt{\theta}/\delta)_r}\ \frac{(W_a\sqrt{\theta}/\delta)_r\ \delta_0}{A_n\ q} - C_{D,n}} \qquad \text{(D29)}$$

where δ_0 and δ are the ratios of the ambient static pressure and the compressor-inlet total pressure, respectively, to NACA standard sea-level static pressure. Because the flight conditions, wing loading, and engine parameters are all known or assumed for any particular power plant and flight plan all the quantities in equations (D29) and (D26) are determined.

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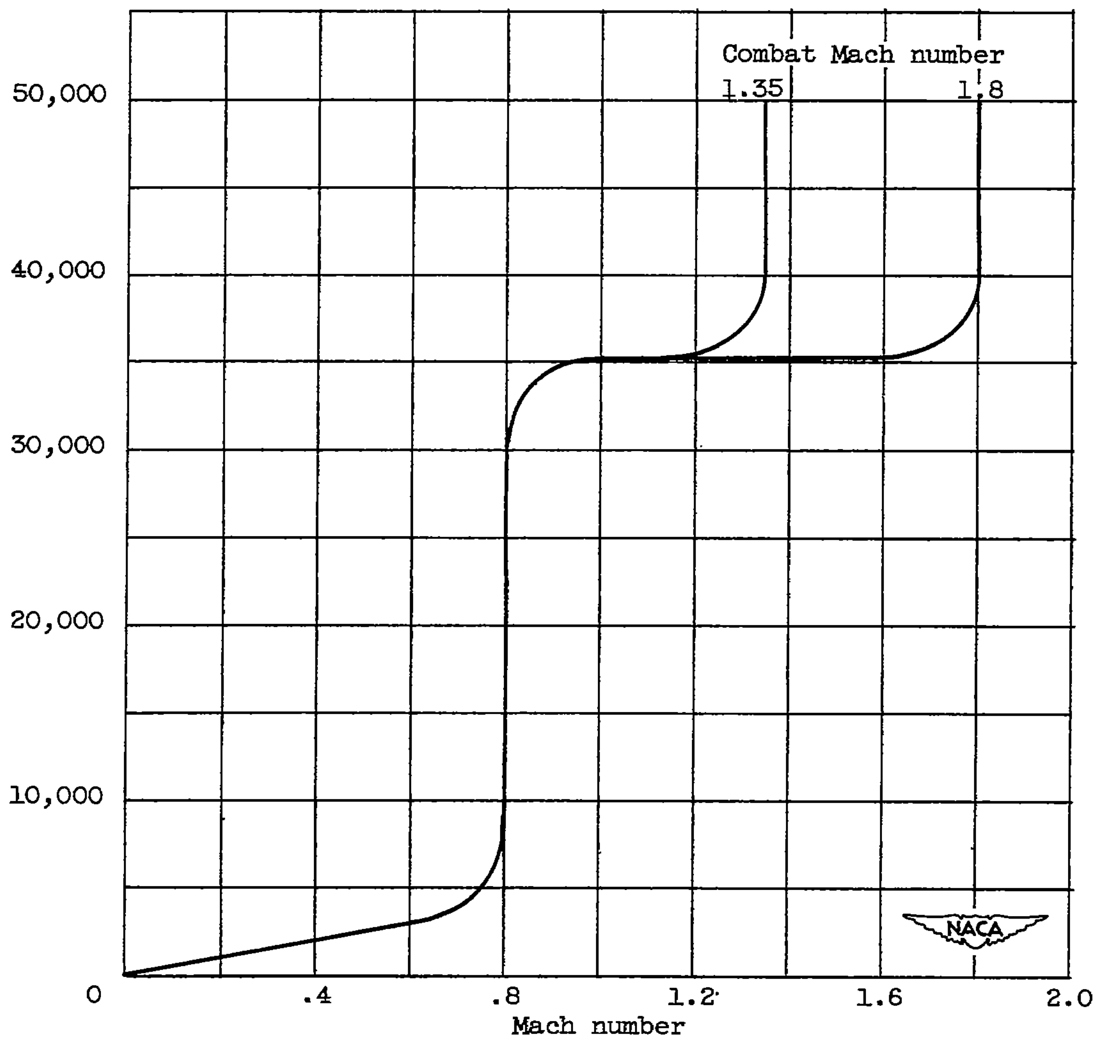

Figure 1. - Flight plan for supersonic interceptor without cruise or loiter provisions.

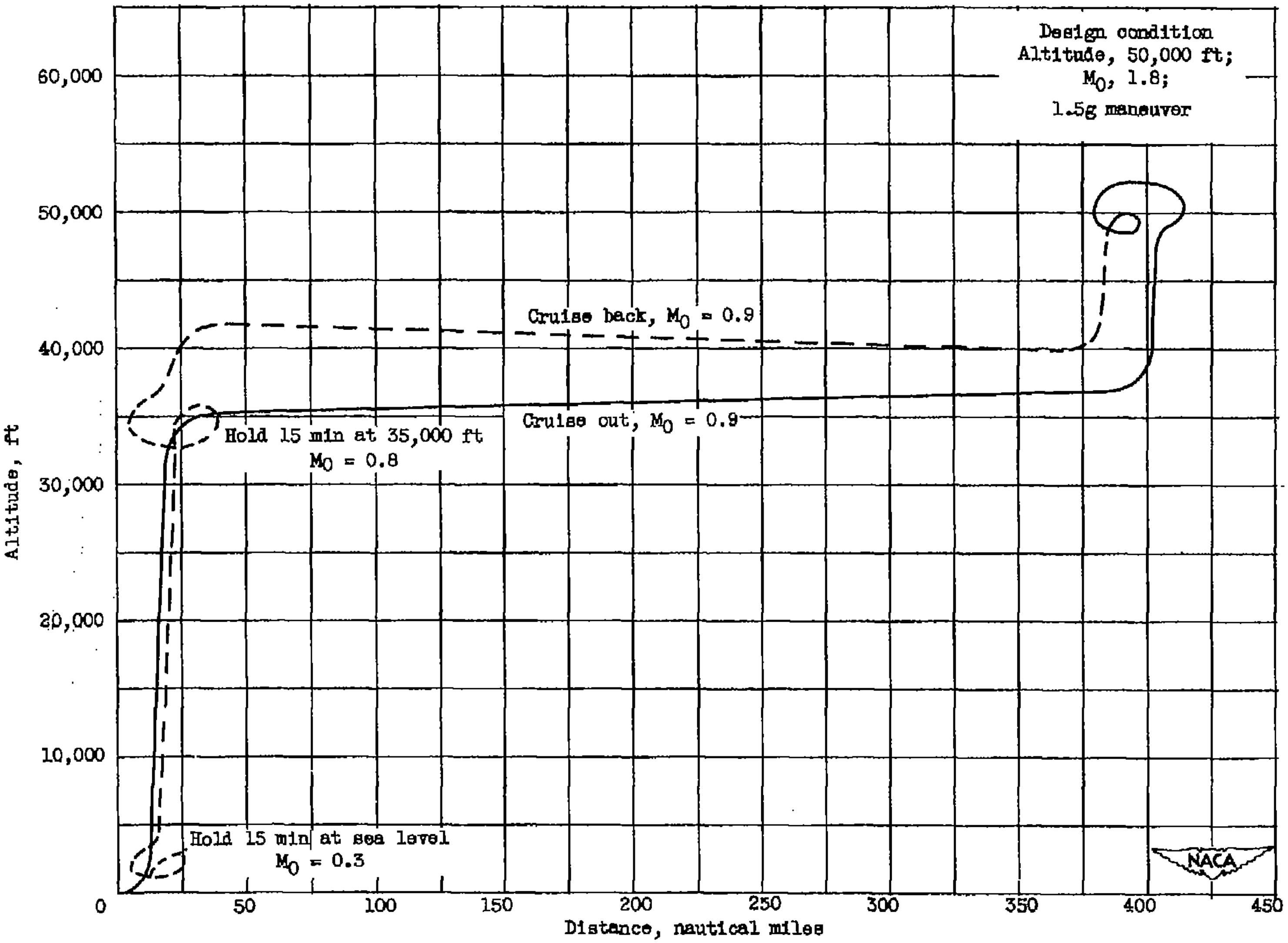

Figure 2. - Flight plan for supersonic interceptor including cruise and loiter provisions.

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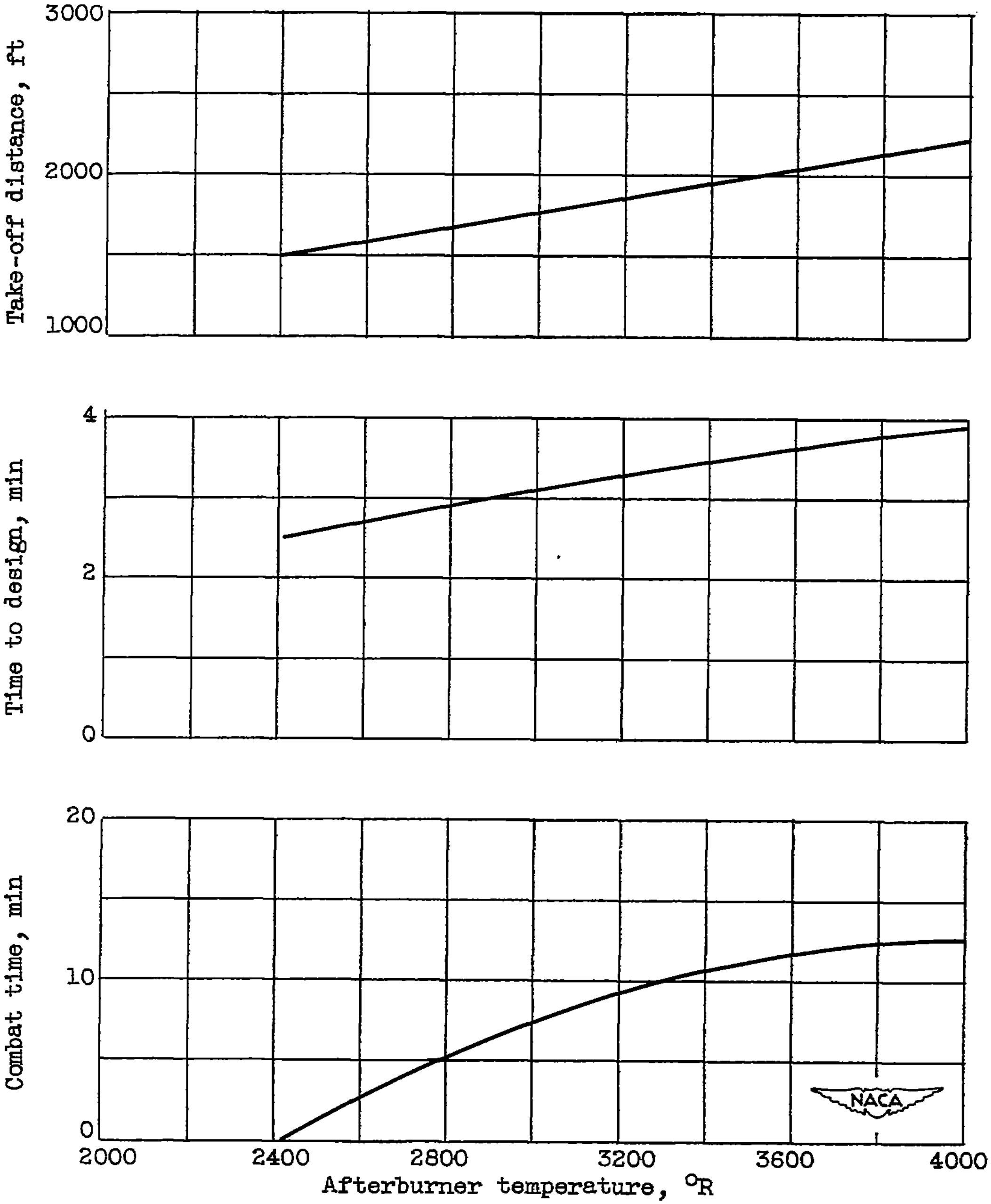

Figure 3. - Effect of afterburner temperature on performance of interceptor without cruise or loiter provisions. Compressor pressure ratio, 5; compressor efficiency, 0.85; turbine-inlet temperature, 2000° R; air flow per unit compressor area, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot; convergent nozzle. Design Mach number, 1.8.

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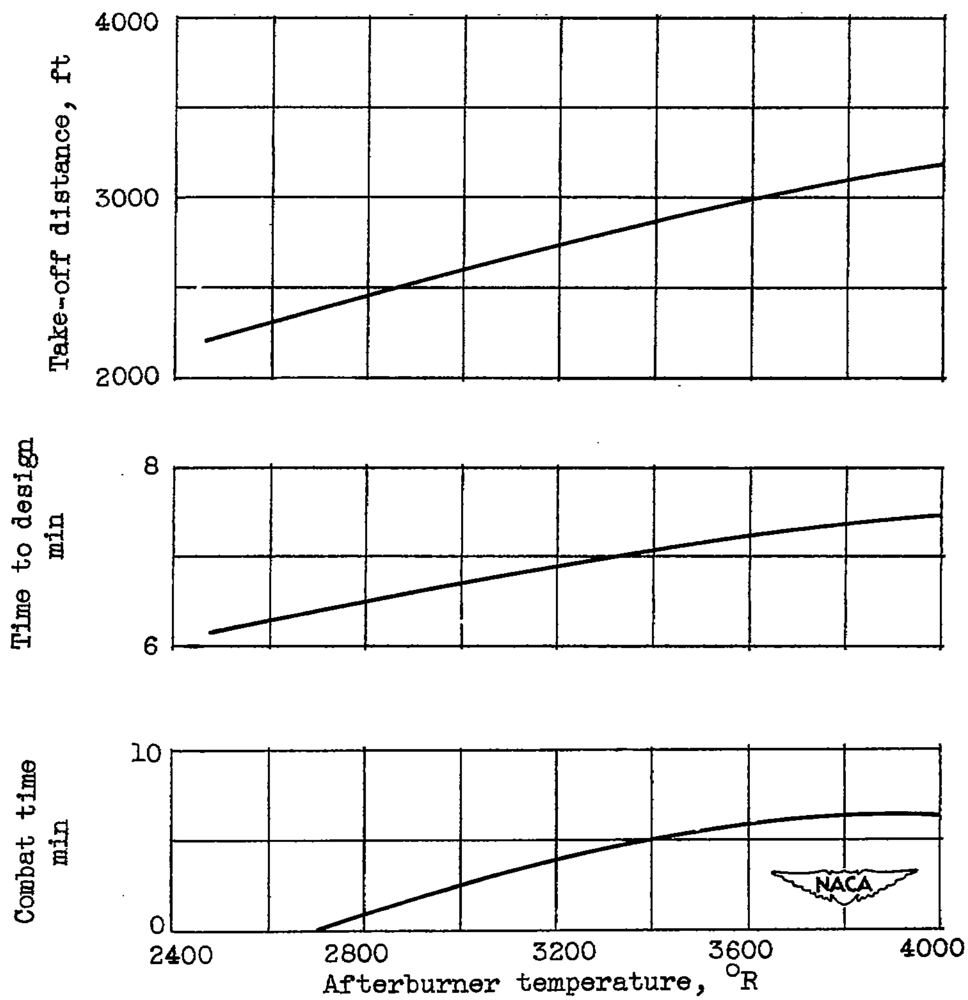

Figure 4. - Effect of afterburner temperature on performance of interceptor with cruise and loiter provisions. Compressor pressure ratio, 7; compressor efficiency, 0.85; turbine-inlet temperature, 2000° R; air flow per unit compressor area, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot; convergent nozzle. Design Mach number, 1.8.

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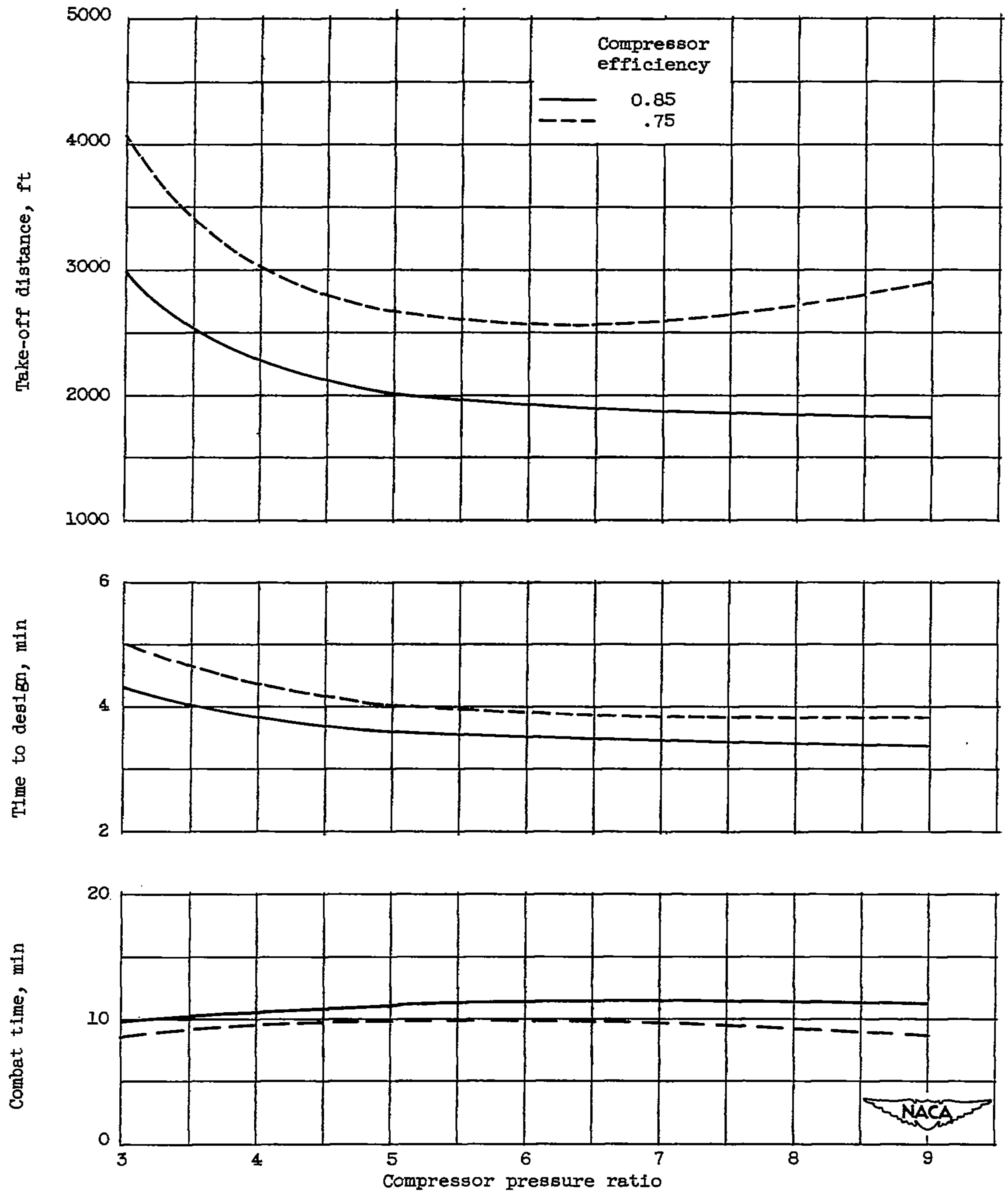

Figure 5. - Effect of compressor pressure ratio and compressor efficiency on performance of interceptor without cruise or loiter provisions. Turbine-inlet temperature, 2000° R; afterburner temperature, 3500° R; air flow per unit compressor area, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot; convergent nozzle. Design Mach number, 1.8.

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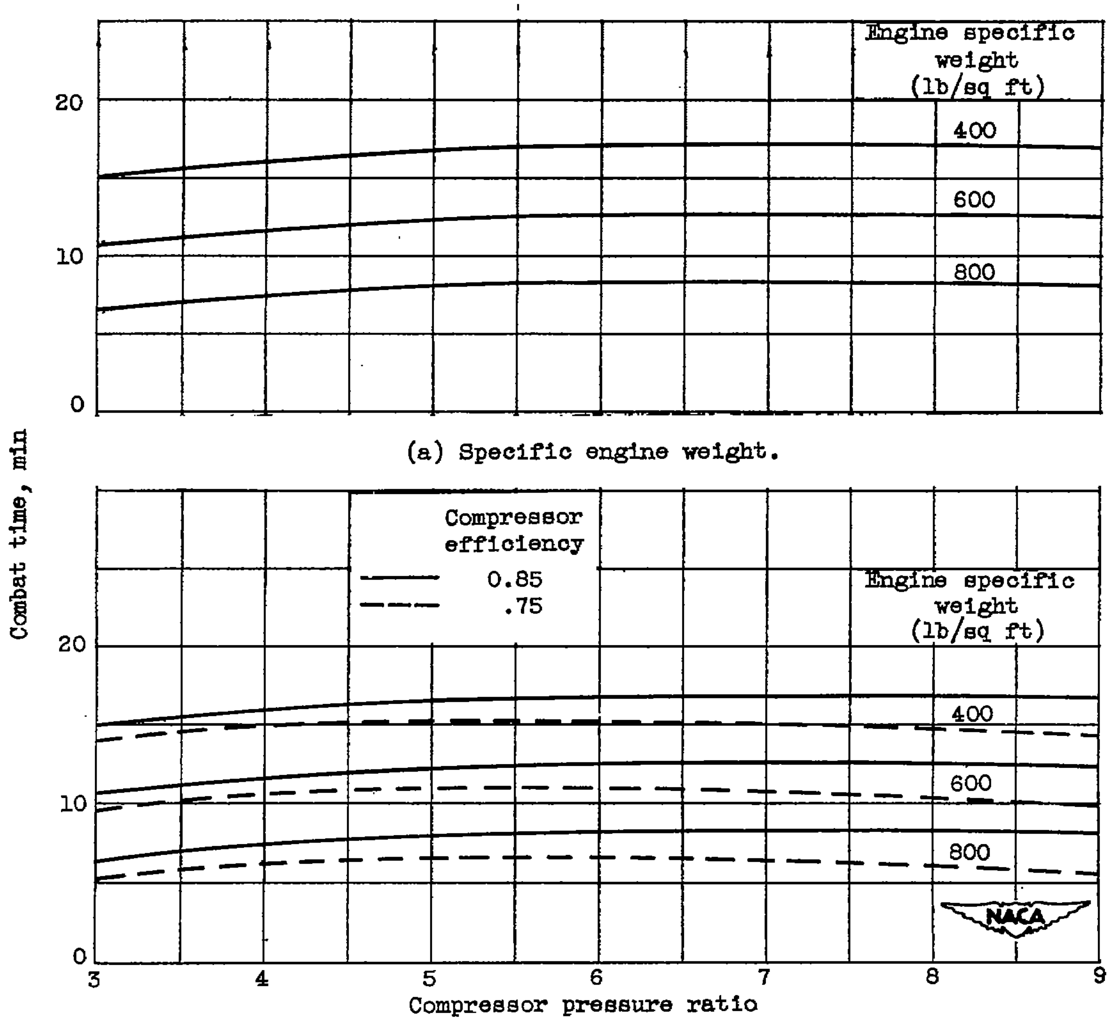

(b) Engine specific weight and compressor efficiency.

Figure 6. - Effect of engine specific weight and compressor efficiency on combat time of interceptor without cruise or loiter provisions for range of compressor pressure ratios. Turbine-inlet temperature, 2000° R; afterburner temperature, 3500° R; air flow per unit compressor area, 27 pounds per second per square foot; convergent nozzle. Design Mach number, 1.8.

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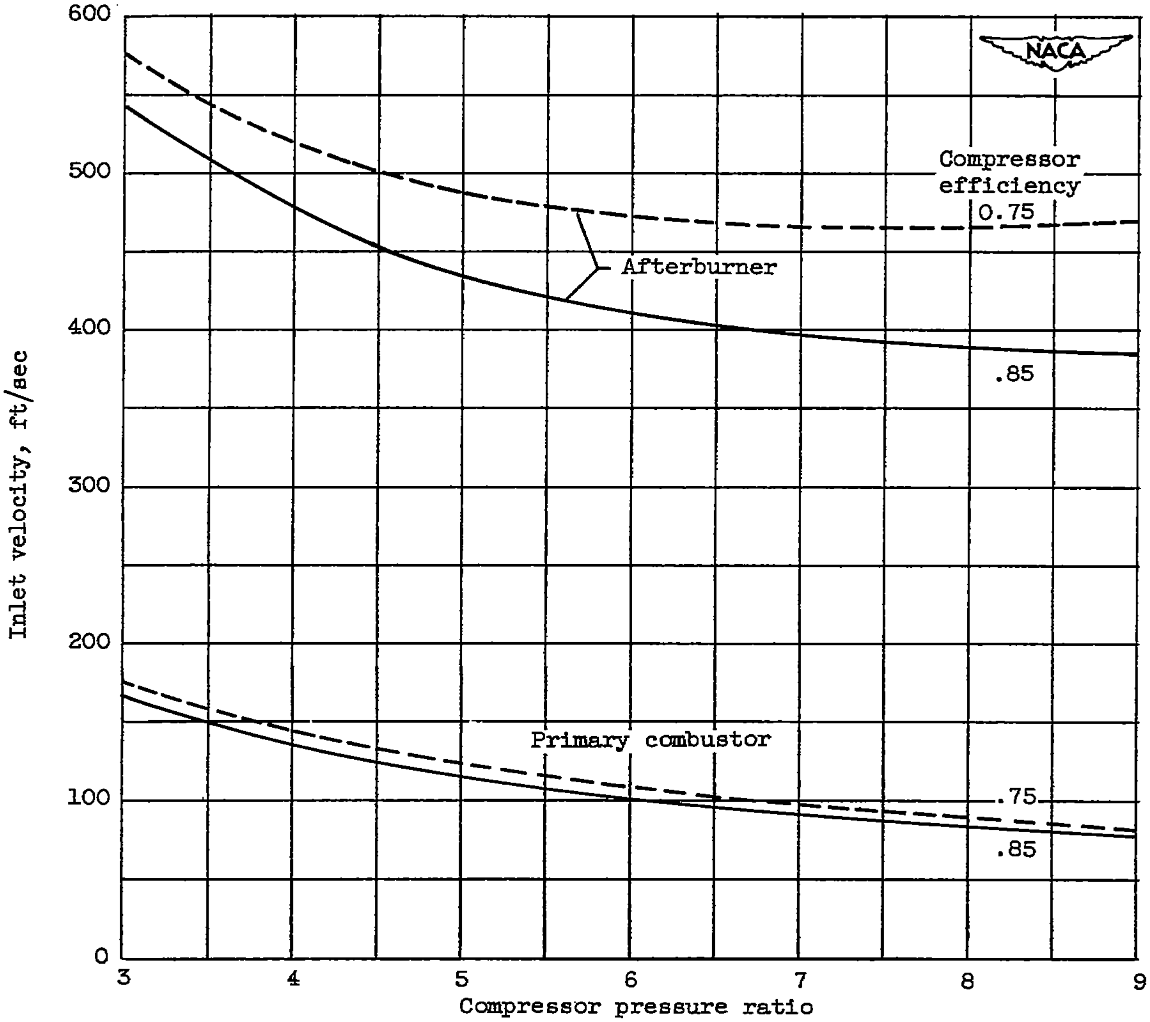

Figure 7. - Effect of compressor pressure ratio and compressor efficiency on burner velocities. Turbine-inlet temperature, 2000° R; air flow per unit compressor area, 27 pounds per second per square foot. Design Mach number, 1.8.

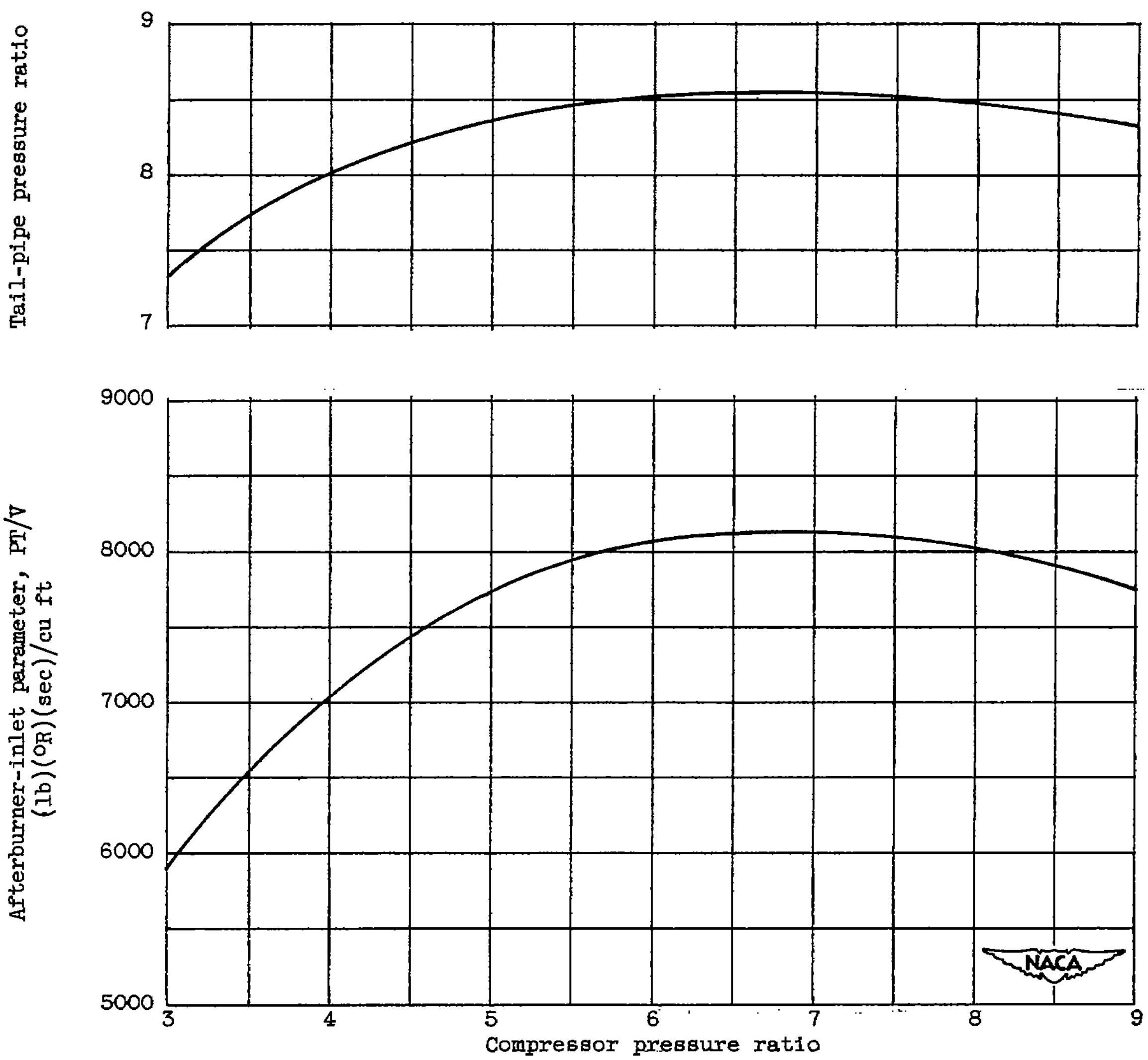

Figure 8. - Effect of compressor pressure ratio on afterburner inlet conditions. Compressor efficiency, 0.85; turbine-inlet temperature, 2000° R; air flow per unit compressor area, 27 pounds per second per square foot. Design Mach number, 1.8.

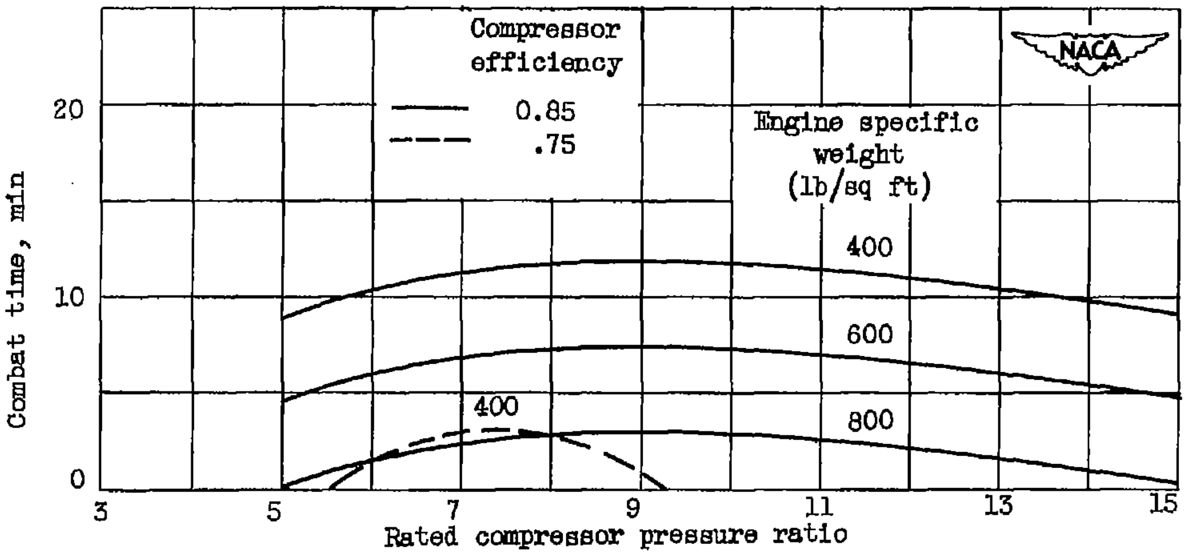

Figure 9. - Effect of compressor efficiency and engine specific weight on combat time of interceptor with cruise and loiter provisions for a range of compressor pressure ratios. Turbine-inlet temperature, 2000° R; afterburner temperature, 3500° R; air flow per unit compressor area, 27 pounds per second per square foot; convergent nozzle. Design Mach number, 1.8.

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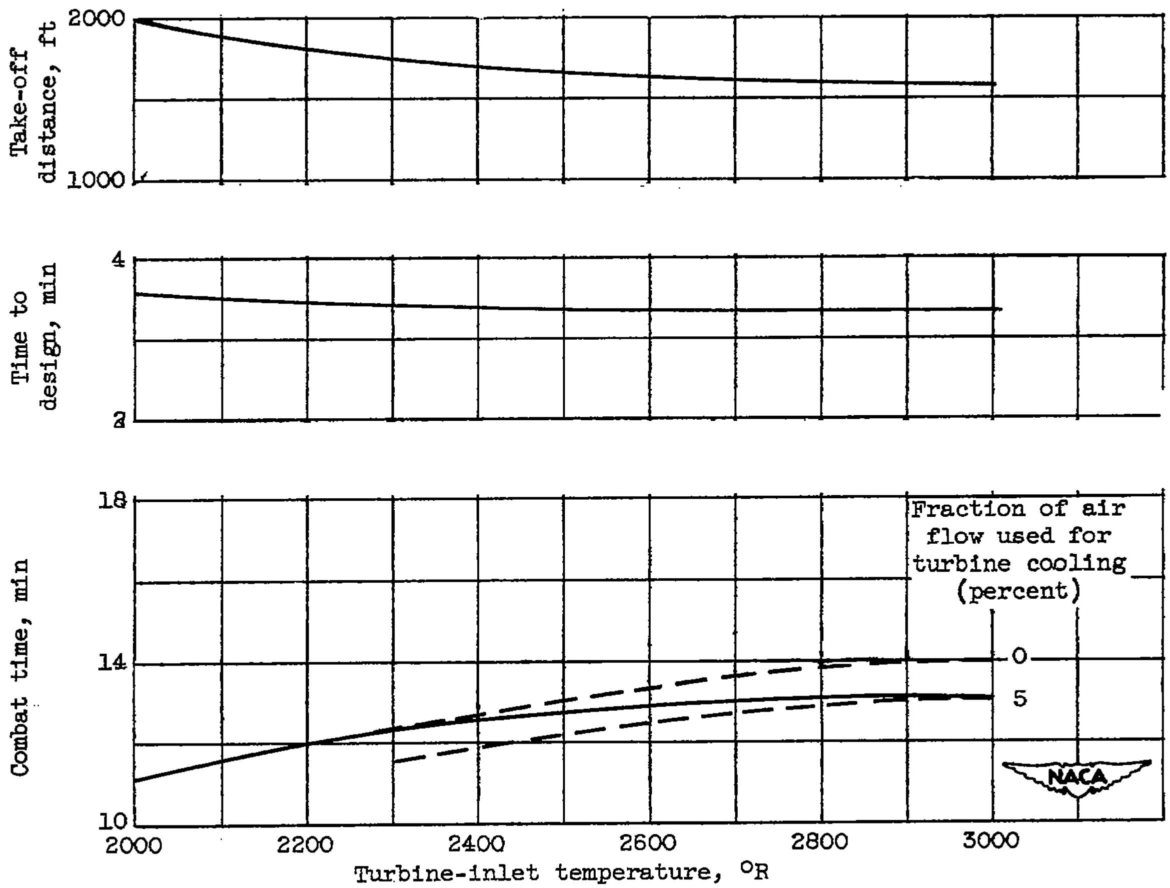

Figure 10. - Effect of turbine-inlet temperature on performance of interceptor without cruise or loiter provisions. Compressor pressure ratio, 5; compressor efficiency, 0.85; afterburner temperature, 3500° R; air flow per unit compressor area, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot; convergent nozzle. Design Mach number, 1.8.

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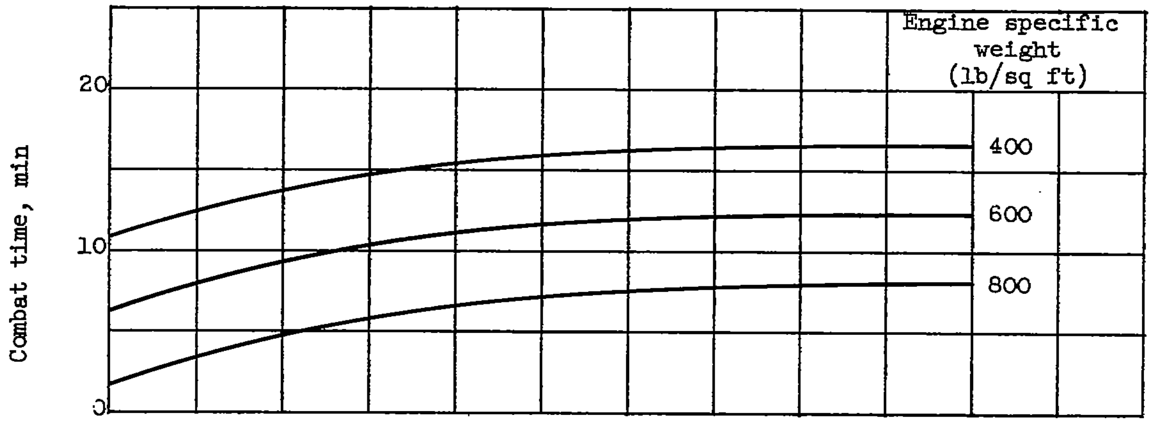
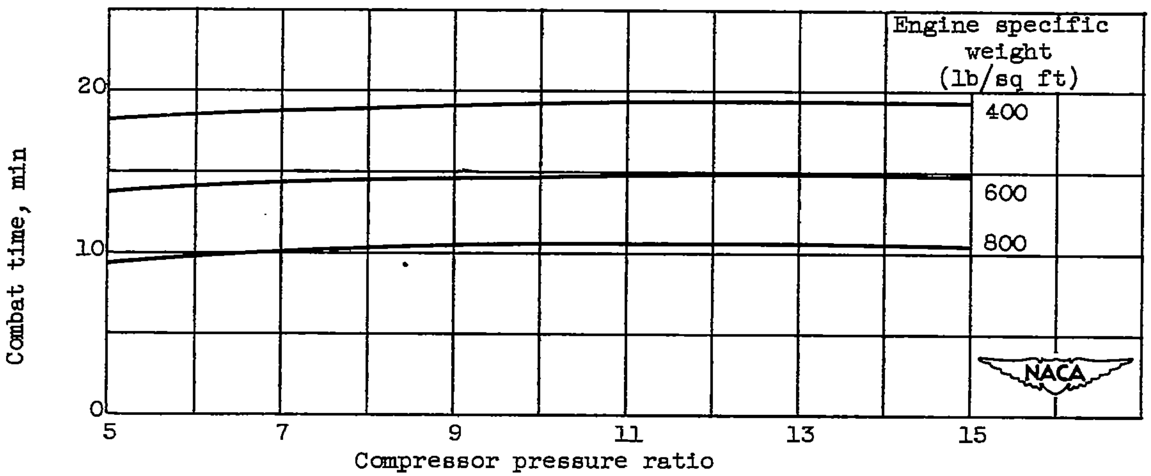


(a) Interceptor with cruise and loiter provisions.



(b) Interceptor without cruise or loiter provisions.

Figure 11. - Effect of compressor pressure ratio and engine specific weight on airplane performance for elevated turbine-inlet temperature. Turbine-inlet temperature, 2500° R; compressor efficiency, 0.85; afterburner temperature, 3500° R; air flow per unit compressor area, 27 pounds per second per square foot; convergent nozzle. Design Mach number, 1.8.

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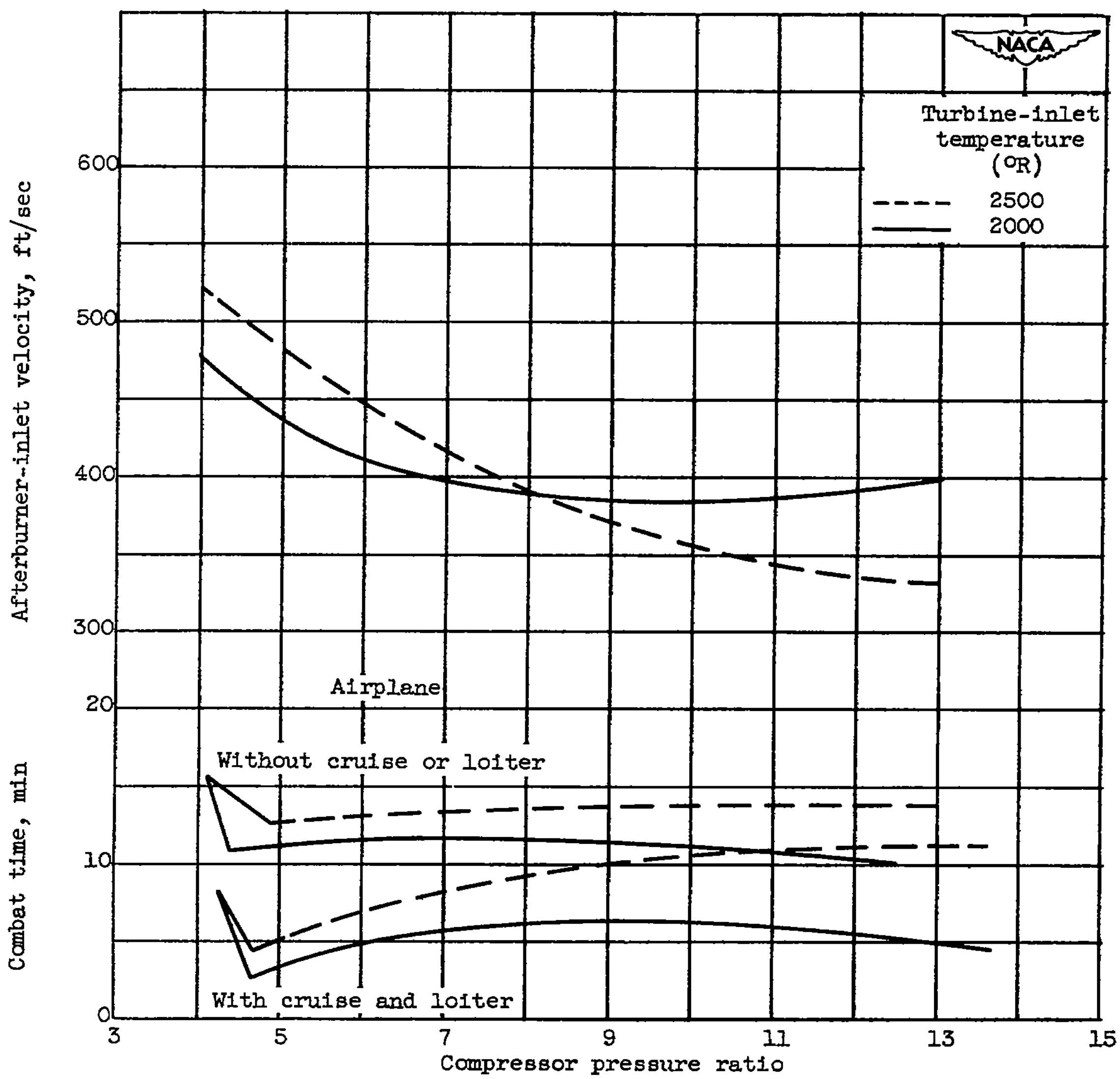

Figure 12. - Effect of turbine-inlet temperature and compressor pressure ratio on performance of both long- and short-range airplanes. Compressor efficiency, 0.85; afterburner temperature, 3500° R; air flow per unit compressor area, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot; convergent nozzle. Design Mach number, 1.8.

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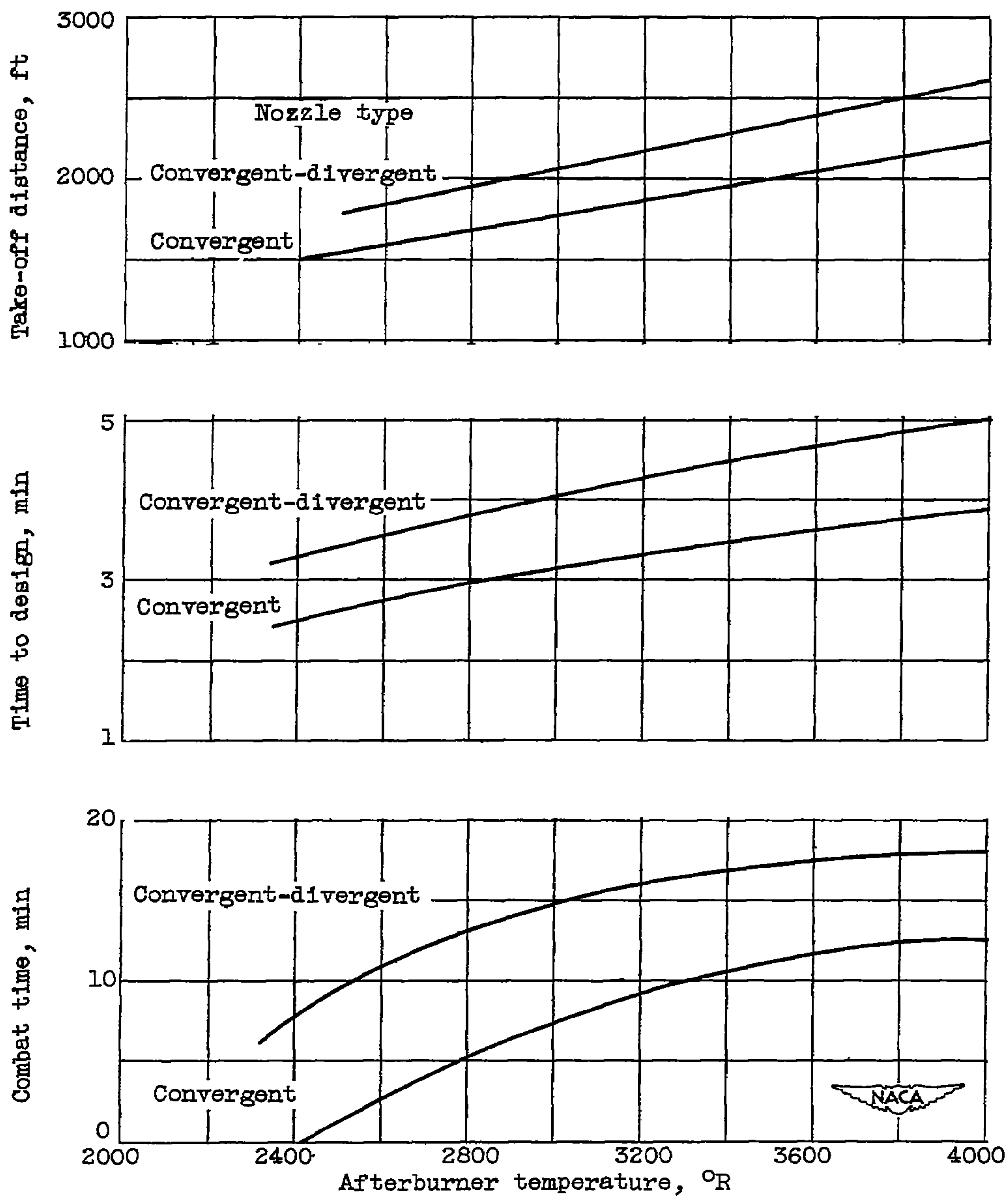

Figure 13. - Comparison of performance of interceptor without cruise or loiter provisions equipped with convergent and convergent-divergent exhaust nozzles. Compressor pressure ratio, 5; compressor efficiency, 0.85; turbine-inlet temperature, 2000° R; afterburner temperature, 3500° R; air flow per unit compressor area, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot. Design Mach number, 1.8.

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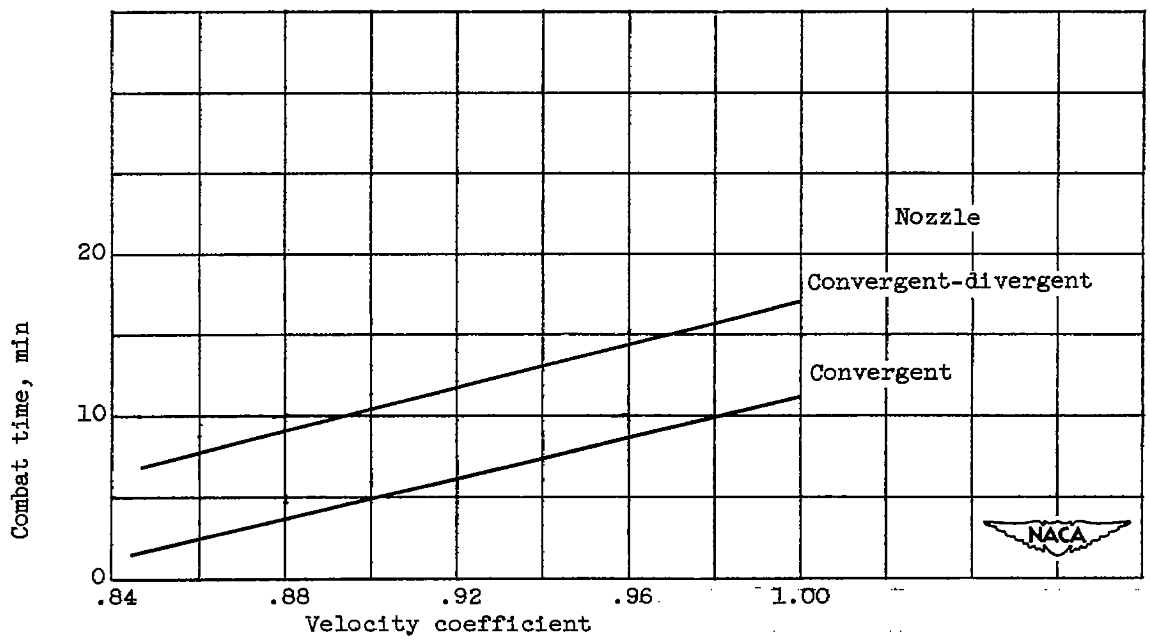

Figure 14. - Effect of exhaust-nozzle velocity coefficient on combat time of interceptor without cruise or loiter provisions. Compressor pressure ratio, 5; compressor efficiency, 0.85; turbine-inlet temperature, 2000° R; afterburner temperature, 3500° R; air flow per unit compressor area, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot. Design Mach number, 1.8.

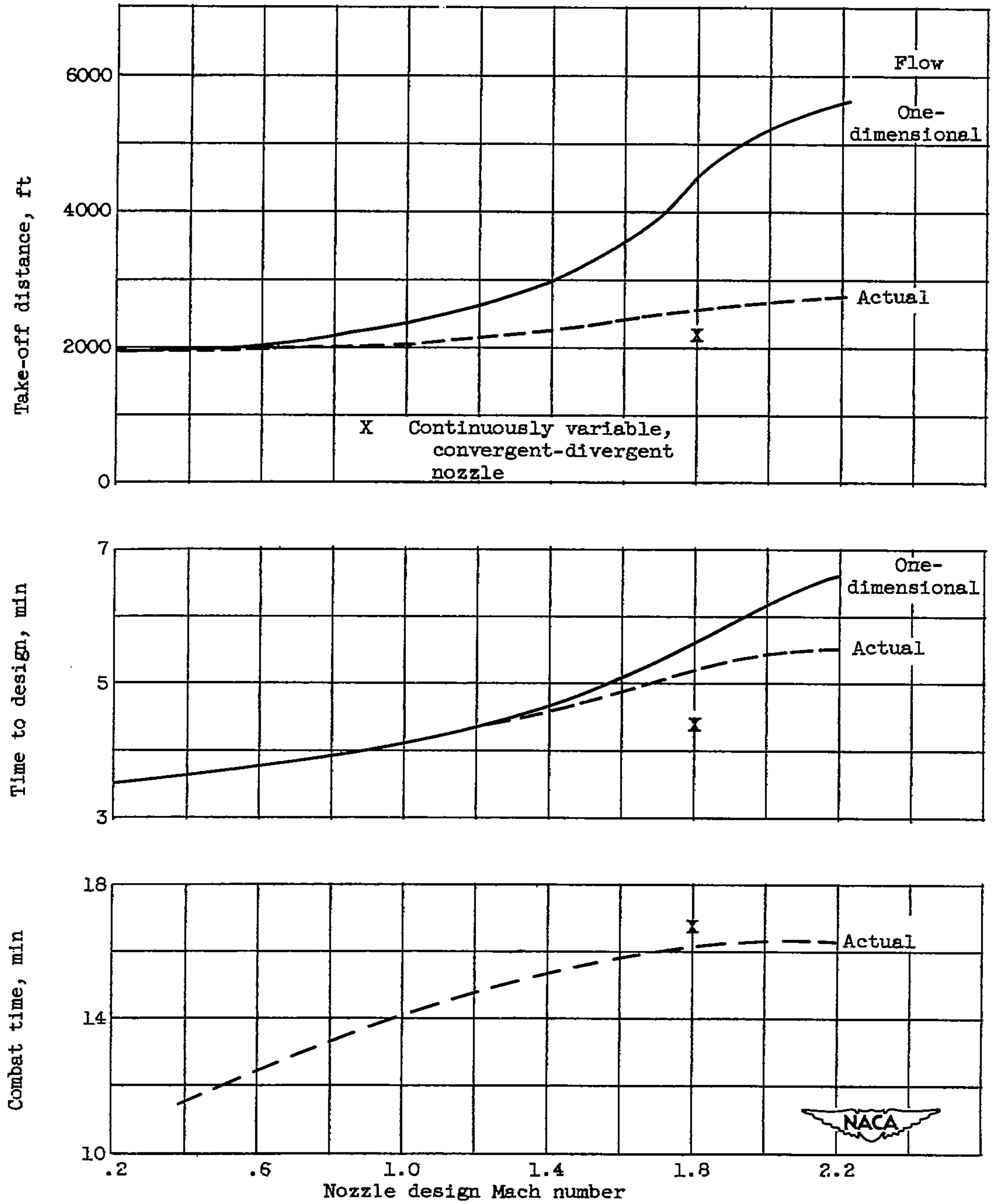

Figure 15. - Effect of exhaust-nozzle configuration on performance of interceptor without cruise or loiter provisions. Compressor pressure ratio, 5; compressor efficiency, 0.85; turbine-inlet temperature, 2000° R; afterburner temperature, 3500° R; air flow per unit compressor area, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot. Design Mach number, 1.8.

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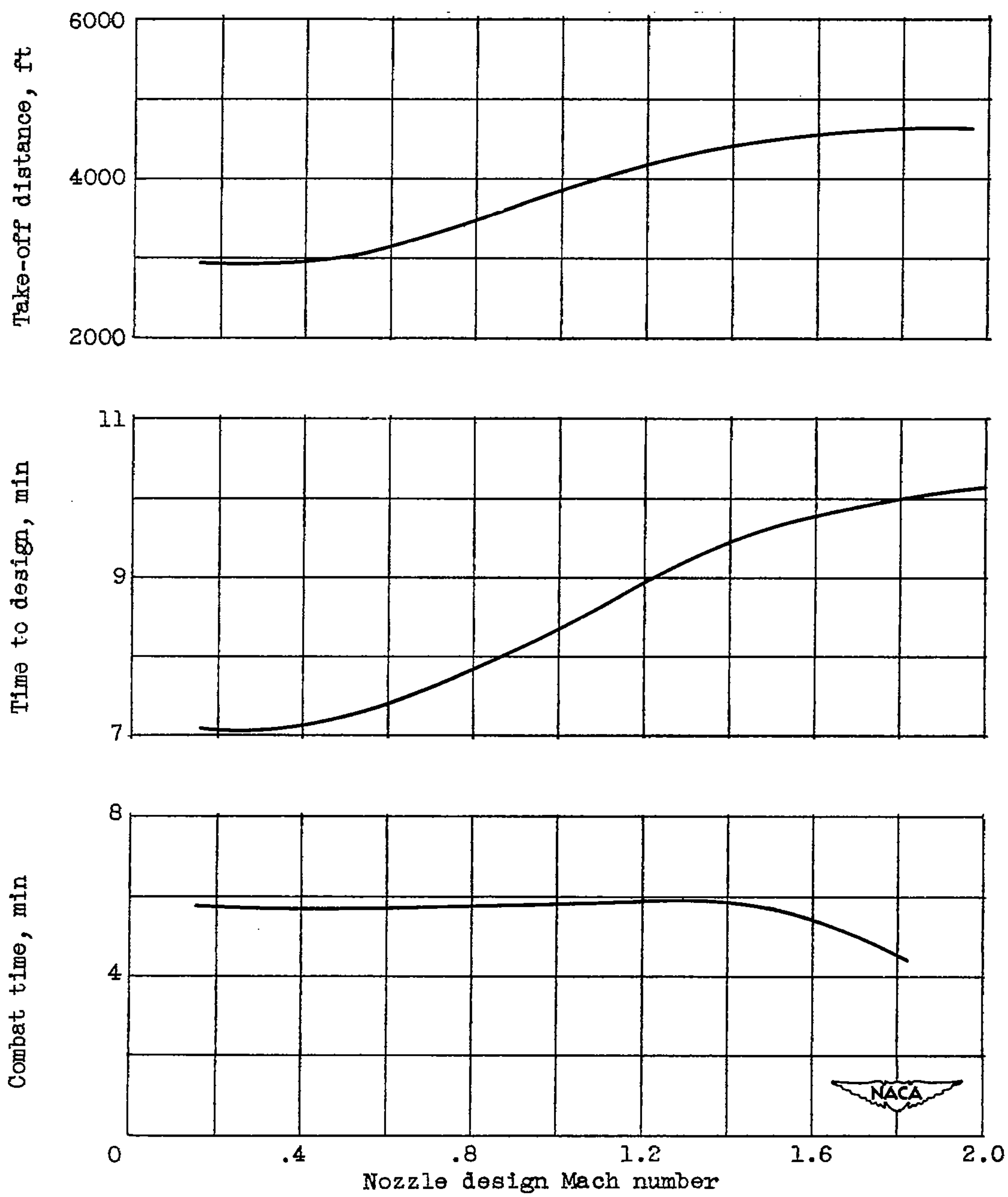

Figure 16. - Effect of convergent-divergent nozzle design Mach number on performance of interceptor including cruise and loiter provisions. Compressor pressure ratio, 7; compressor efficiency, 0.85; turbine-inlet temperature, 2000° R; afterburner temperature, 3500° R; air flow per unit compressor area, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot. Design Mach number, 1.8.

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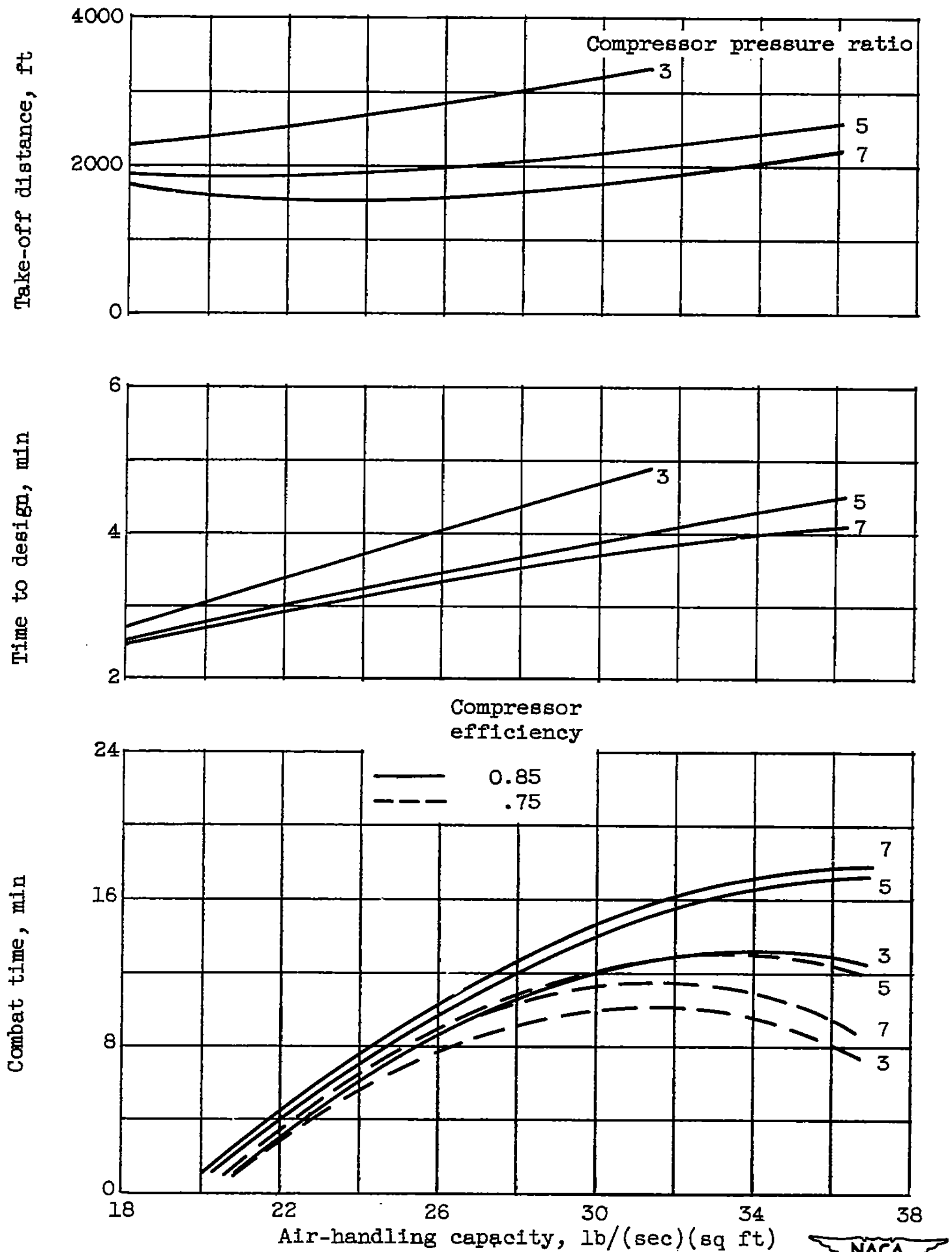

Figure 17. - Effect of air-handling capacity and compressor pressure ratio on performance of interceptor without cruise or loiter provisions. Turbine-inlet temperature, 2000° R; afterburner temperature, 3500° R; engine specific weight, 650 pounds per square foot; convergent nozzle. Design Mach number, 1.8.

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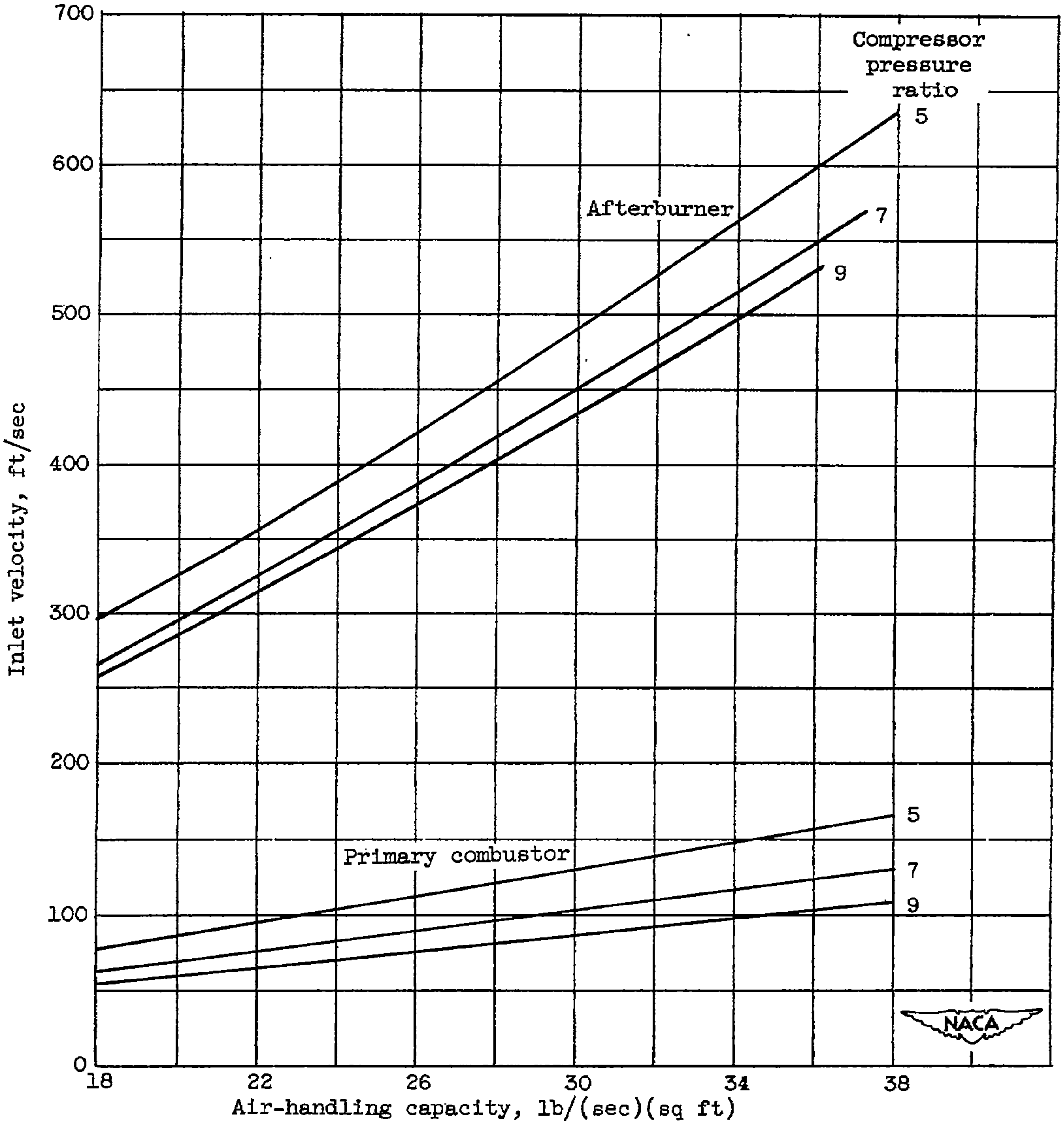

Figure 18. - Effect of air-handling capacity on burner velocities.

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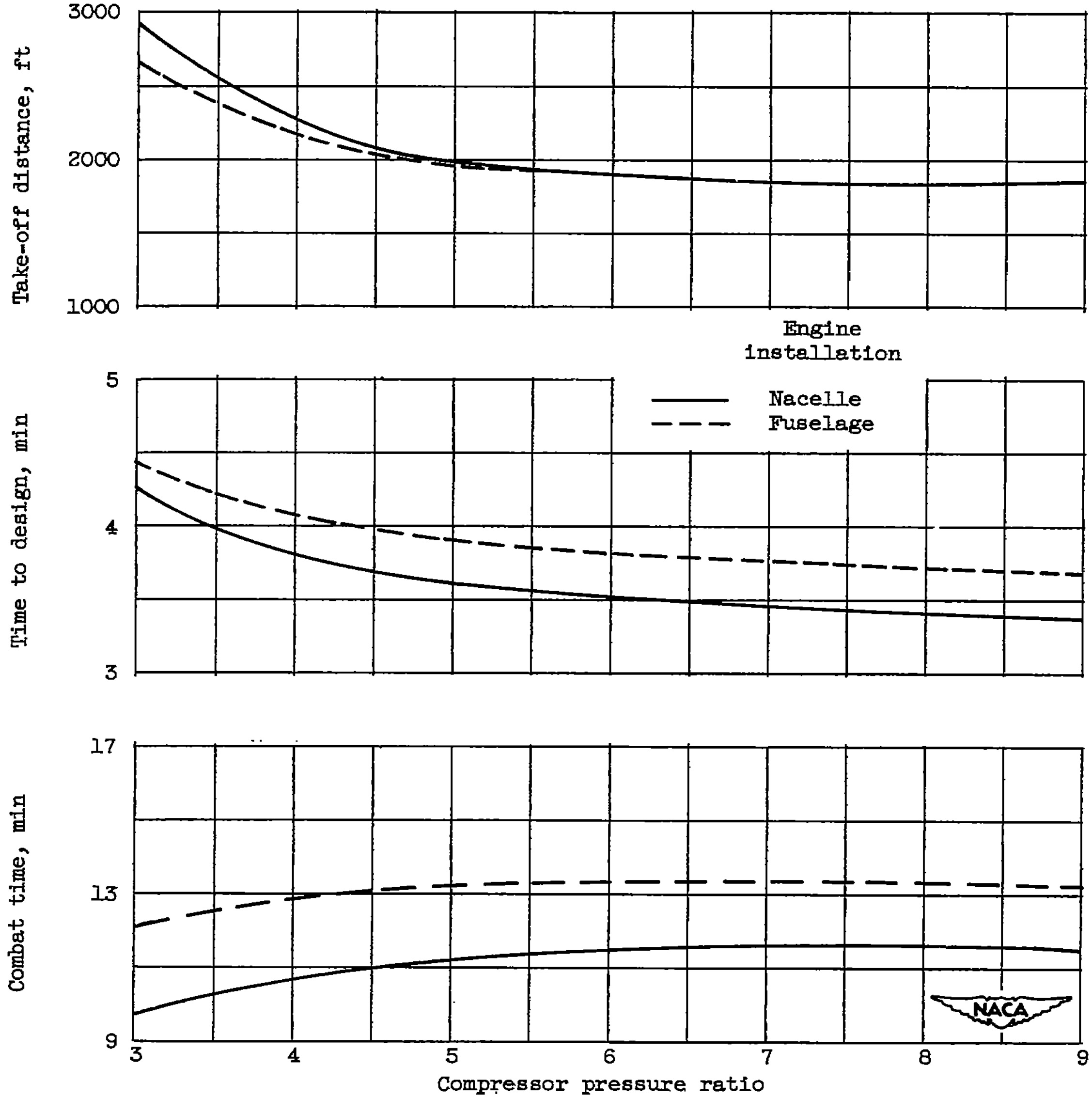

Figure 19. - Comparison of performance of interceptor without cruise or loiter provisions with engines mounted in nacelles and submerged in fuselage. Compressor efficiency, 0.85; turbine-inlet temperature, 2000° R; afterburner temperature, 3500° R; air flow per unit compressor area, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot; convergent nozzle. Design Mach number, 1.8.

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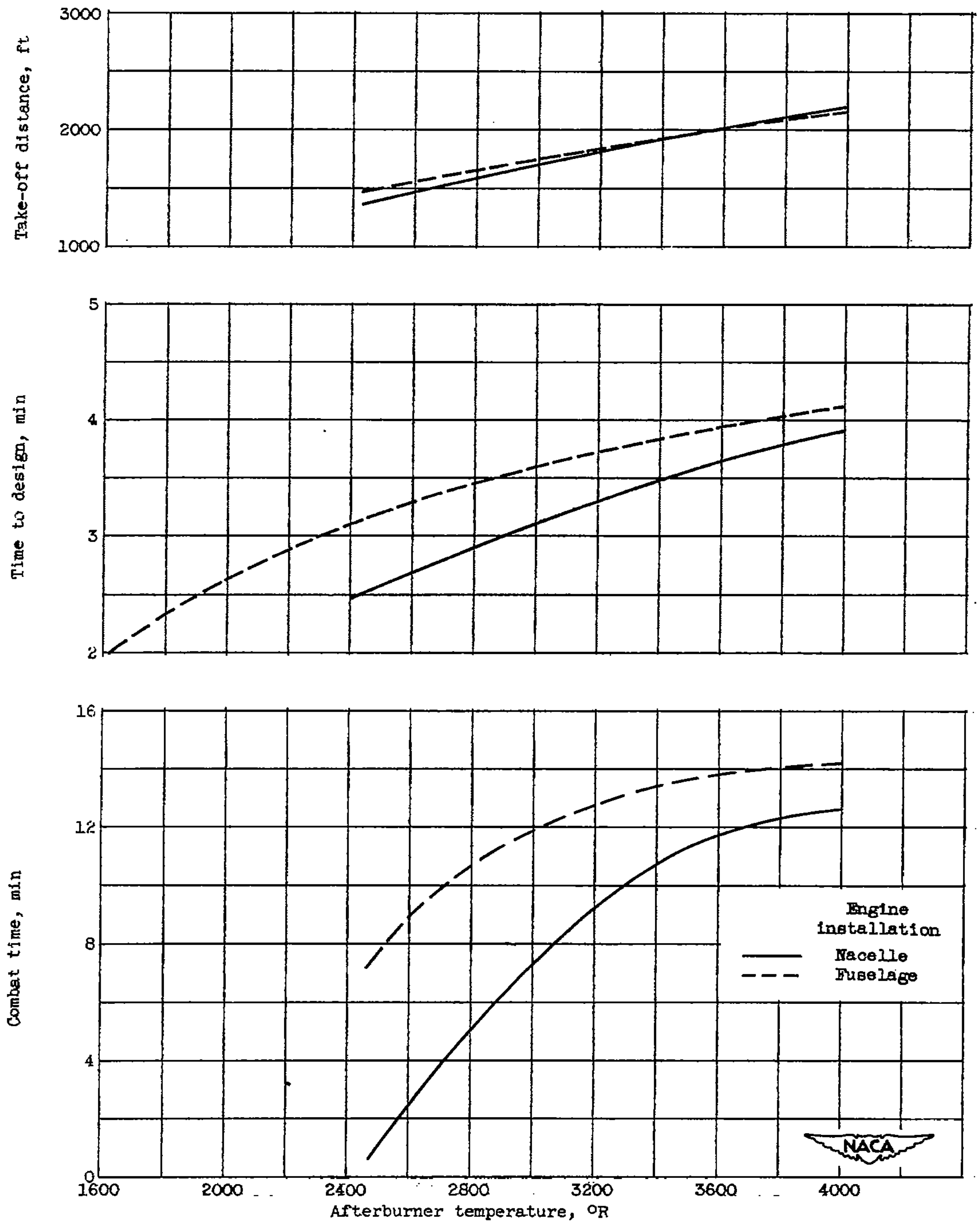

Figure 20. - Effect of afterburner temperature on performance trends of interceptor without cruise or loiter provisions with engines in nacelles and with engine submerged in fuselage. Compressor pressure ratio, 5; compressor efficiency, 0.85; turbine-inlet temperature, 2000° R; air flow per unit compressor area, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot; convergent nozzle. Design Mach number, 1.8.

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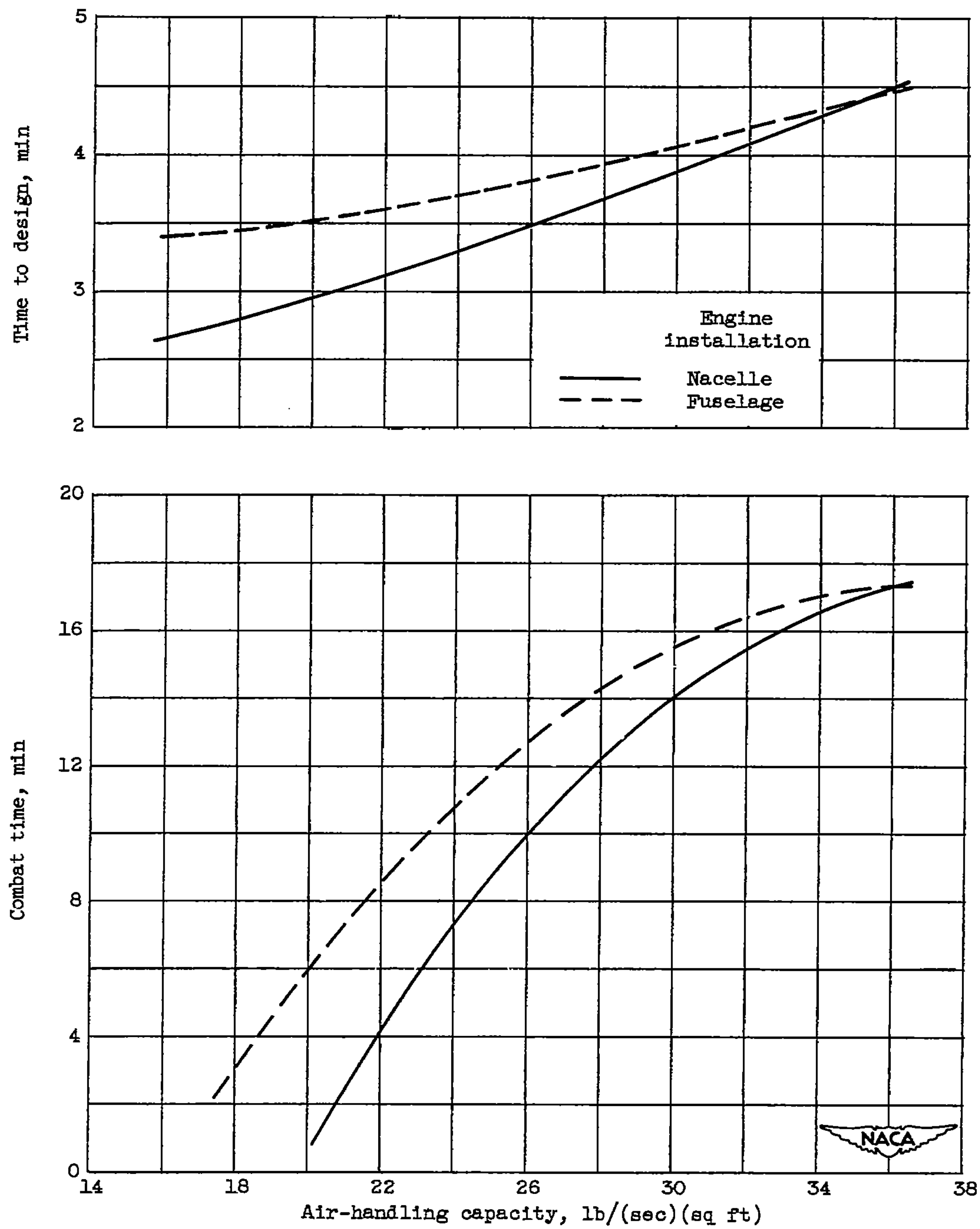

Figure 21. - Effect of air-handling capacity on performance trends of interceptor without cruise or loiter provisions with engines in nacelles and engine submerged in fuselage. Compressor pressure ratio, 5; compressor efficiency, 0.85; turbine-inlet temperature, 2000° R; afterburner temperature, 3500° R; engine specific weight, 650 pounds per square foot; convergent nozzle. Design Mach number, 1.8.

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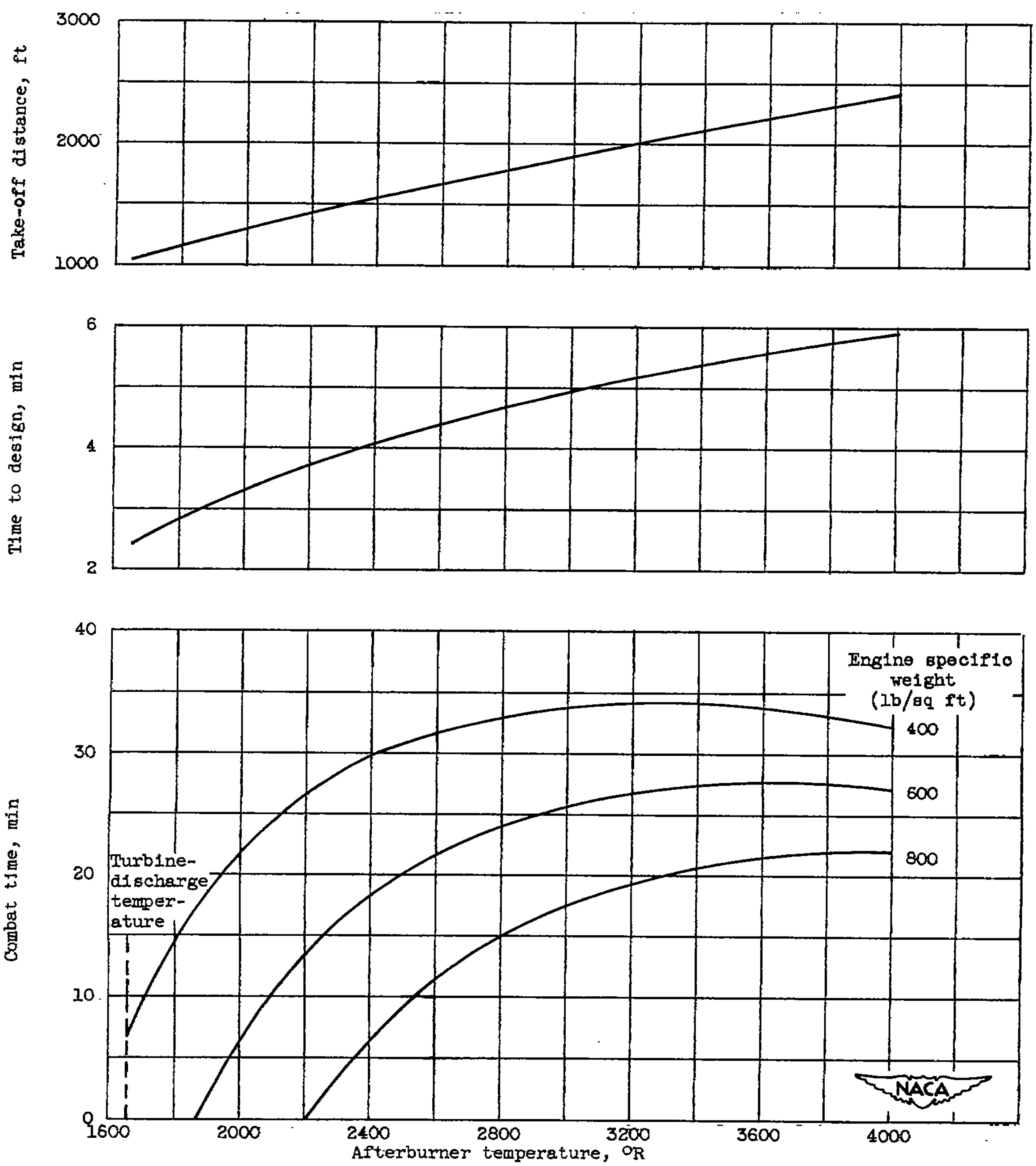

Figure 22. - Effect of afterburner temperature and engine specific weight on performance of interceptor without cruise or loiter provisions. Compressor pressure ratio, 5; compressor efficiency, 0.85; turbine-inlet temperature, 2000° R; air flow per unit compressor area, 27 pounds per second per square foot; convergent nozzle. Design Mach number, 1.35.

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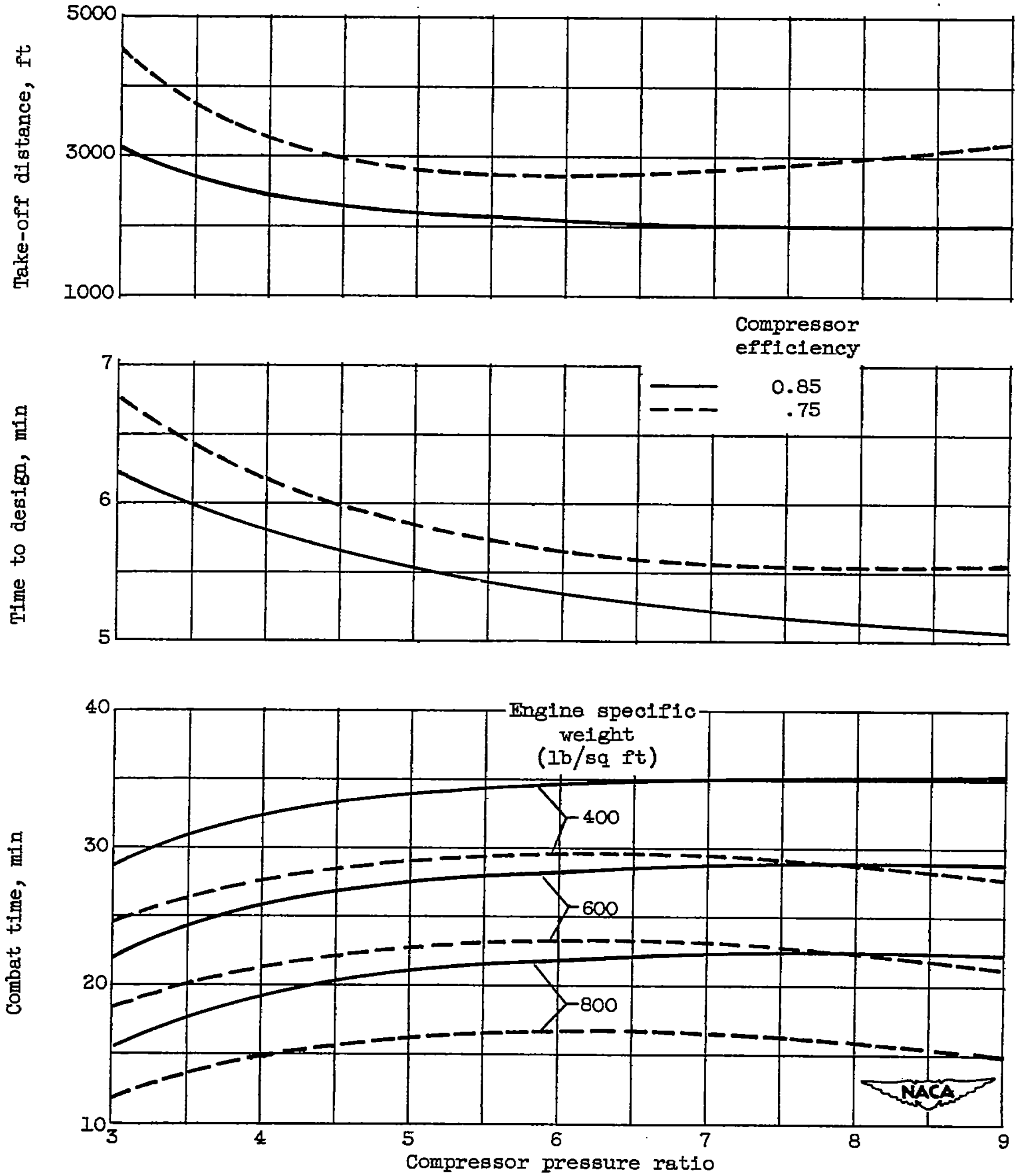

Figure 23. - Effect of compressor pressure ratio, compressor efficiency, and engine specific weight on performance of interceptor without cruise or loiter provisions. Turbine-inlet temperature, 2000° R; afterburner temperature, 3500° R; air flow per unit compressor area, 27 pounds per second per square foot; convergent nozzle. Design Mach number, 1.35.

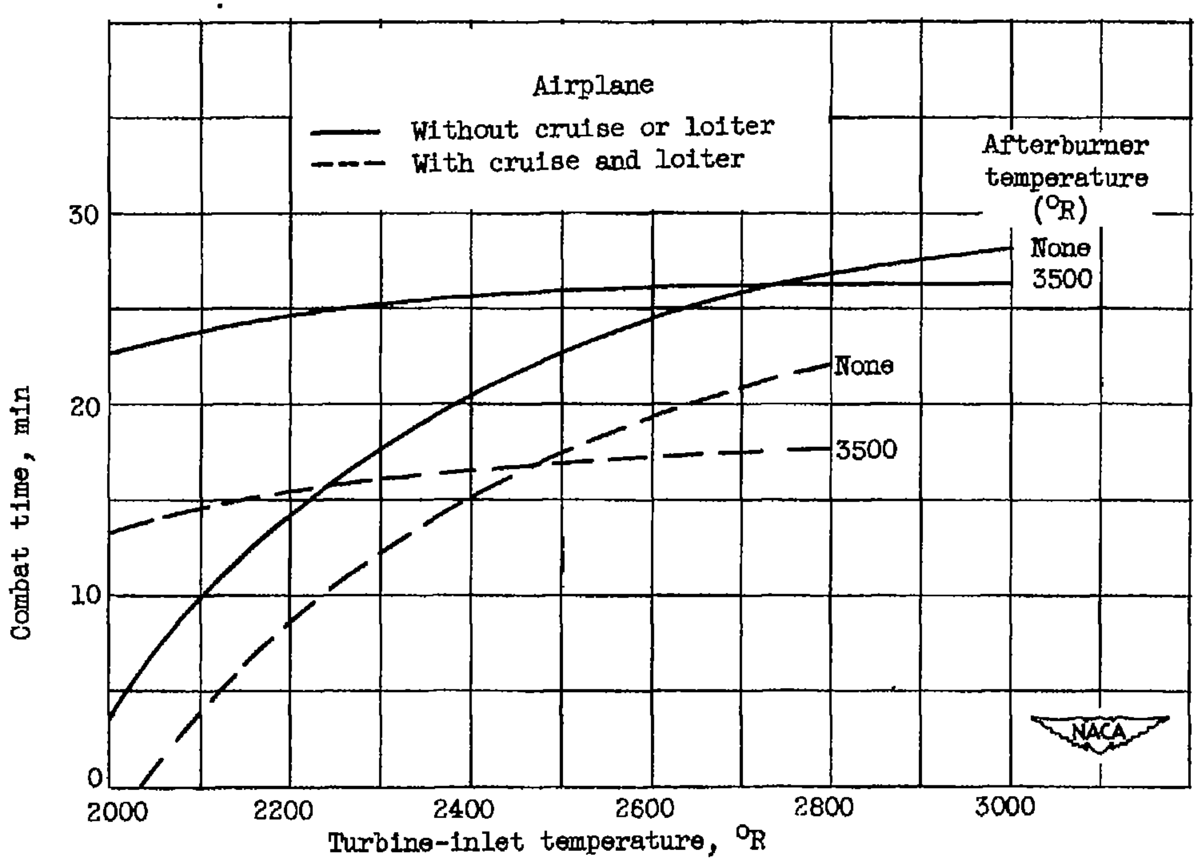

Figure 24. - Comparison of effect of turbine-inlet temperature and afterburner temperature on combat time of two different interceptor flight missions with advanced engine. Compressor pressure ratio, 7; compressor efficiency, 0.85; air flow per unit compressor area, 33 pounds per second per square foot; engine specific weight, 650 pounds per square foot; convergent-divergent nozzle. Design Mach number, 1.8.

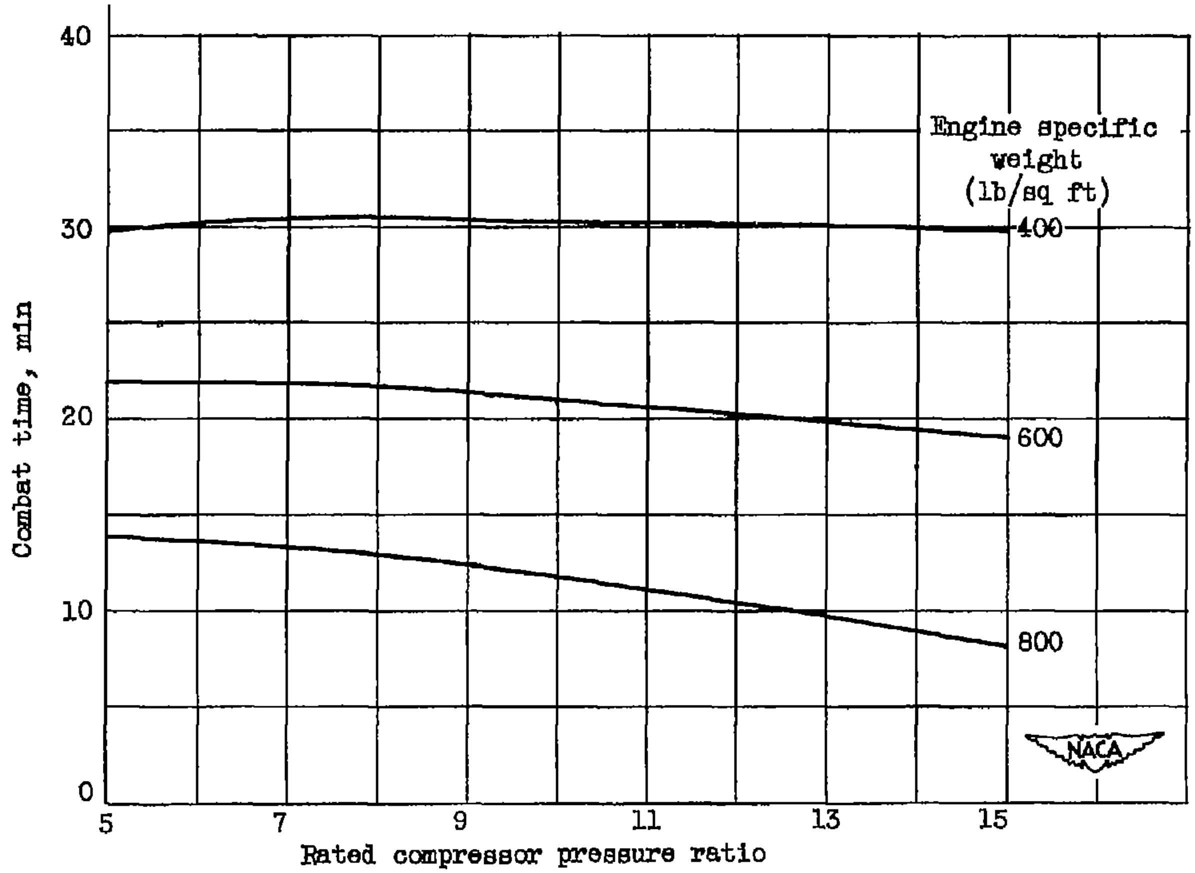

Figure 25. - Effect of compressor pressure ratio on combat time of interceptor without cruise or loiter provisions with nonafterburning, advanced engine. Compressor efficiency, 0.85; turbine-inlet temperature, 2500° R; air flow per unit compressor area, 33 pounds per second per square foot; convergent-divergent nozzle. Design Mach number, 1.8.

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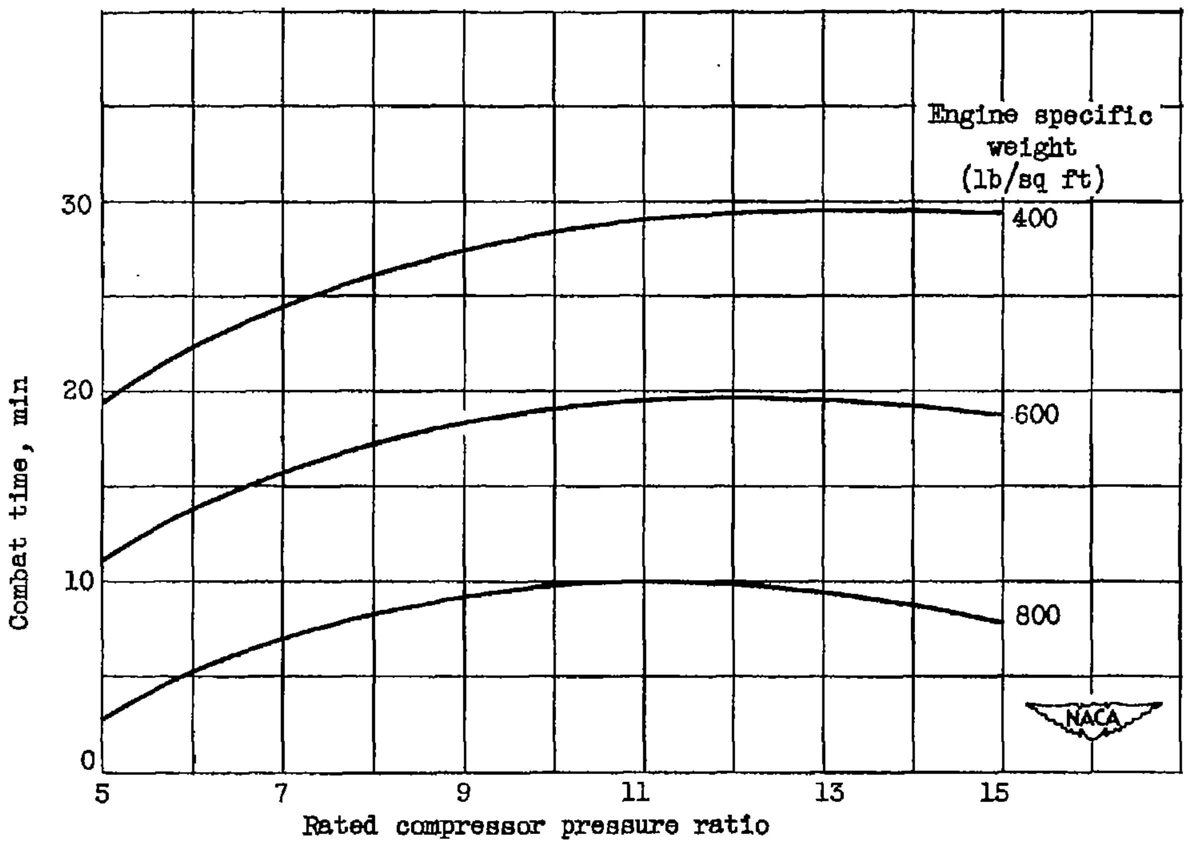

Figure 26. - Effect of compressor pressure ratio on combat time of interceptor with cruise and loiter provisions with nonafterburning, advanced engine. Compressor efficiency, 0.85; turbine-inlet temperature, 2500° R; air flow per unit compressor area, 33 pounds per second per square foot; convergent-divergent nozzle. Design Mach number, 1.8.

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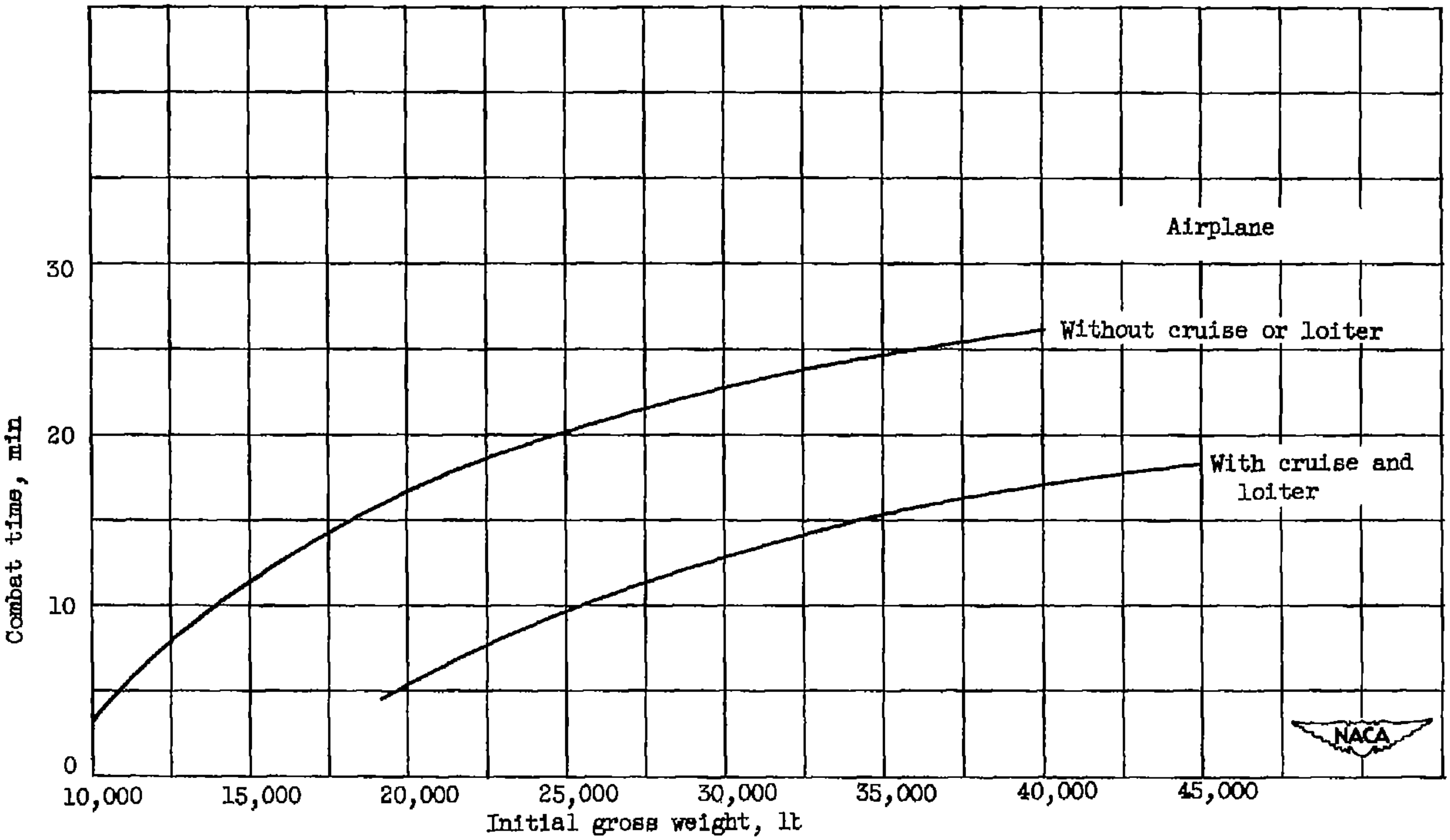

Figure 27. - Effect of initial gross weight on combat time of two different interceptor missions with advanced engines. Compressor pressure ratio, 7; compressor efficiency, 0.85; turbine-inlet temperature, 2500° R; afterburner temperature, 3500° R; air flow per unit compressor area, 33 pounds per second per square foot; engine specific weight, 650 pounds per square foot; convergent-divergent nozzle. Design Mach number, 1.8.

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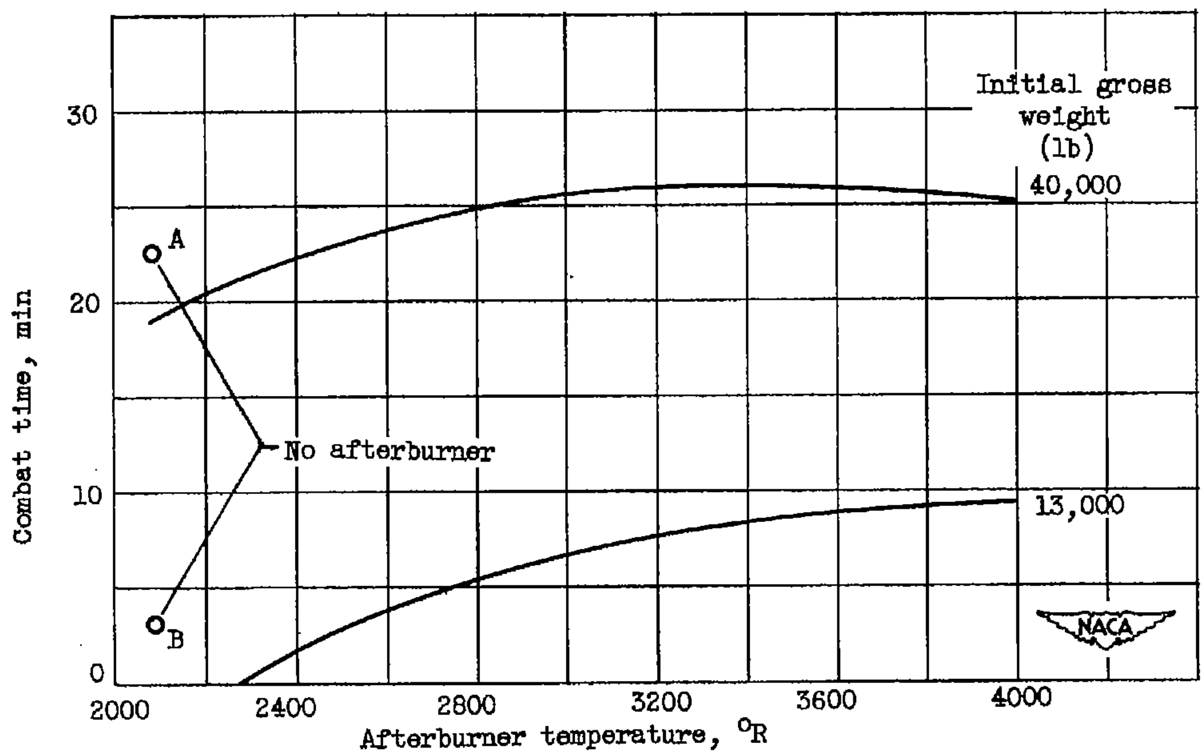

Figure 28. - Effect of initial gross weight on afterburner temperature for maximum combat time of interceptor without cruise or loiter with advanced engine. Compressor pressure ratio, 7; compressor efficiency, 0.85; turbine-inlet temperature, 2500° R; air flow per unit compressor area, 33 pounds per second per square foot; engine specific weight, 650 pounds per square foot; convergent-divergent nozzle. Design Mach number, 1.8.

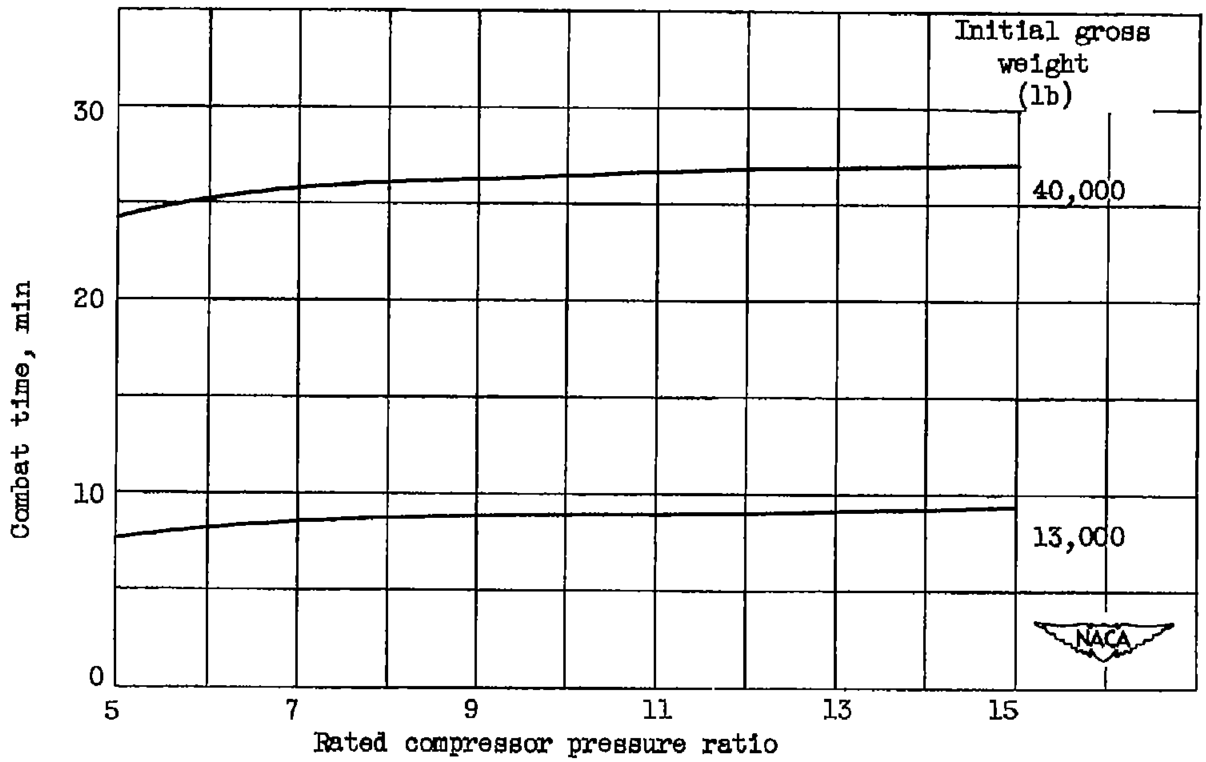

Figure 29. - Effect of initial gross weight on compressor pressure ratio for maximum combat time of interceptor without cruise or loiter with advanced engine. Compressor efficiency, 0.85; turbine-inlet temperature, 2500° R; afterburner temperature, 3500° R; air flow per unit compressor area, 33 pounds per second per square foot; engine specific weight, 650 pounds per square foot; convergent-divergent nozzle. Design Mach number, 1.8.

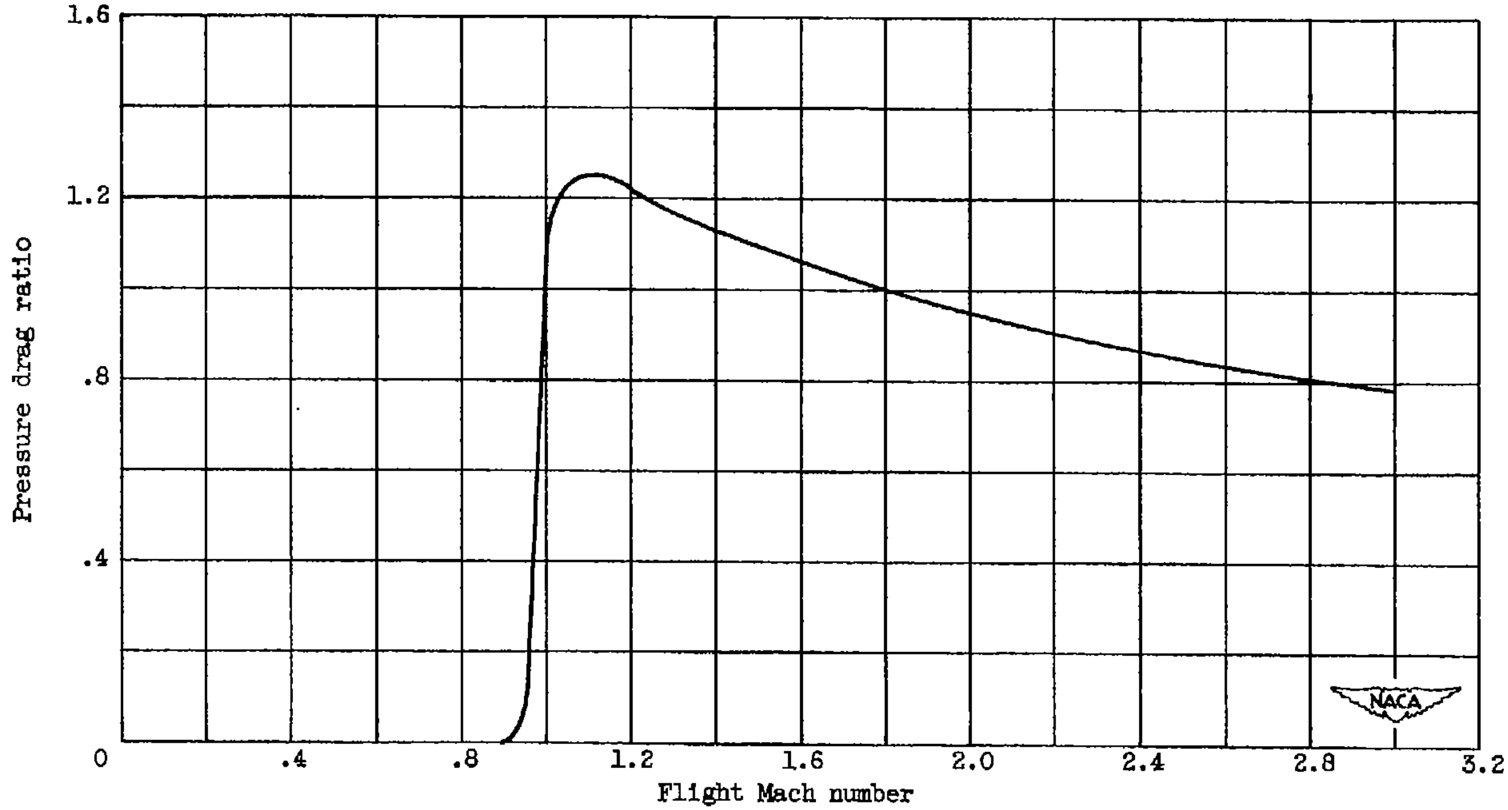

Figure 30. - Assumed pressure drag variation with flight Mach number.

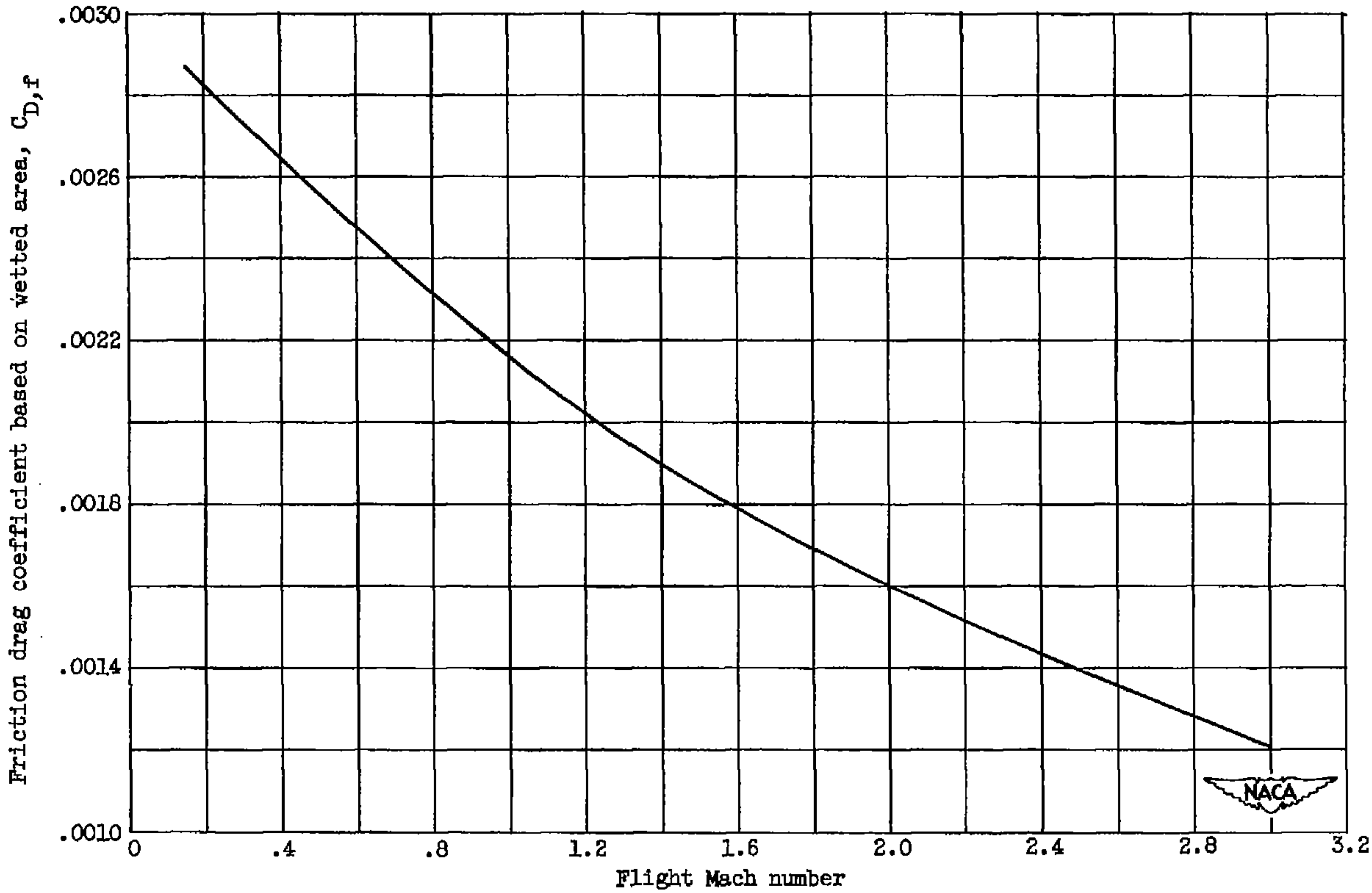

Figure 31. - Assumed variation of friction drag coefficient for fuselage and nacelle based on wetted area with flight Mach number.

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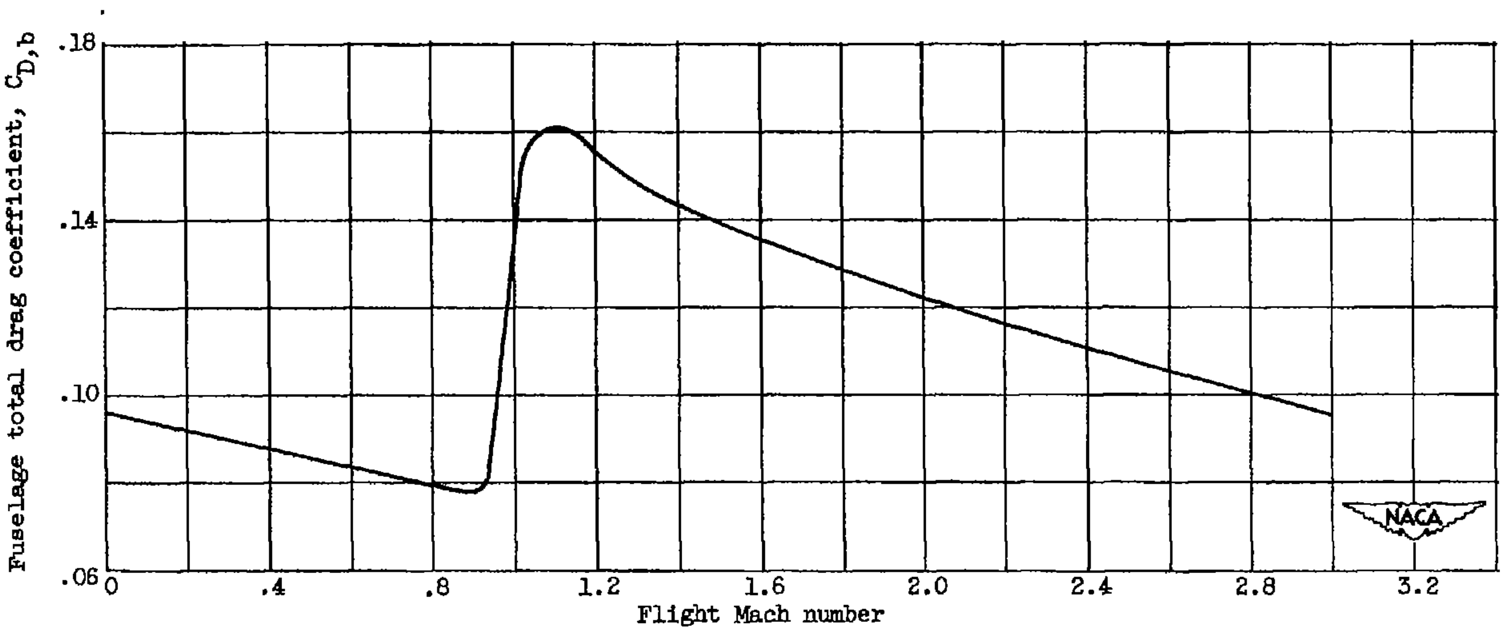

Figure 32. - Assumed fuselage drag coefficient based on fuselage frontal area as function of flight Mach number.

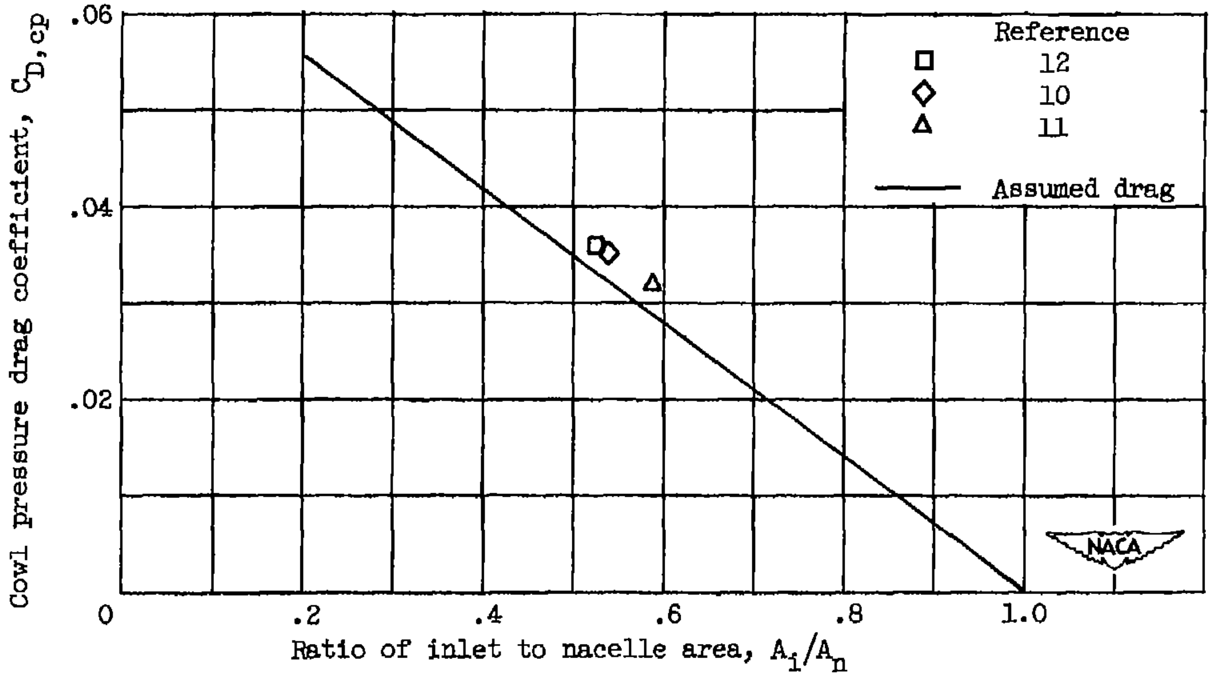

Figure 33. - Assumed variation of cowl pressure drag coefficient based on frontal area as function of ratio of inlet area to frontal area. Mach number, 1.8.

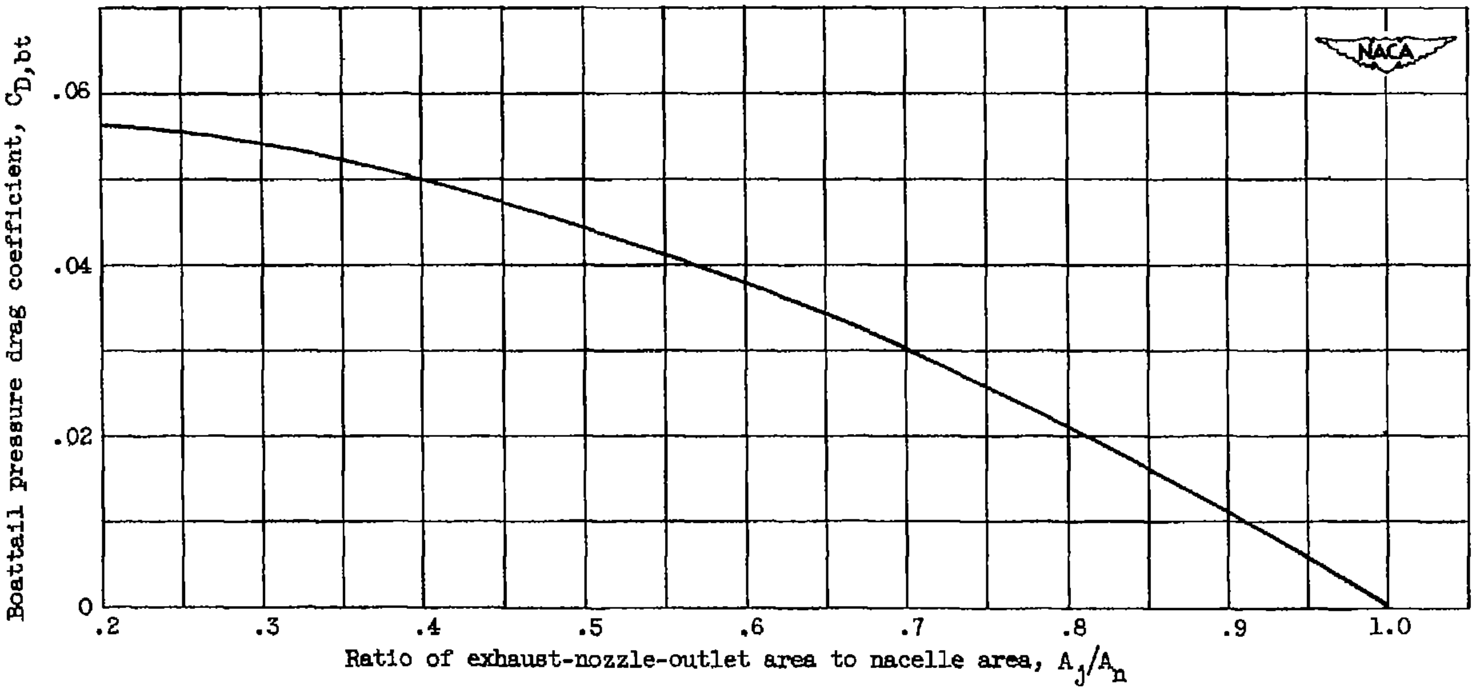

Figure 34. - Assumed variation of boattail pressure drag coefficient with ratio of exhaust-nozzle area to frontal area. Mach number, 1.8.

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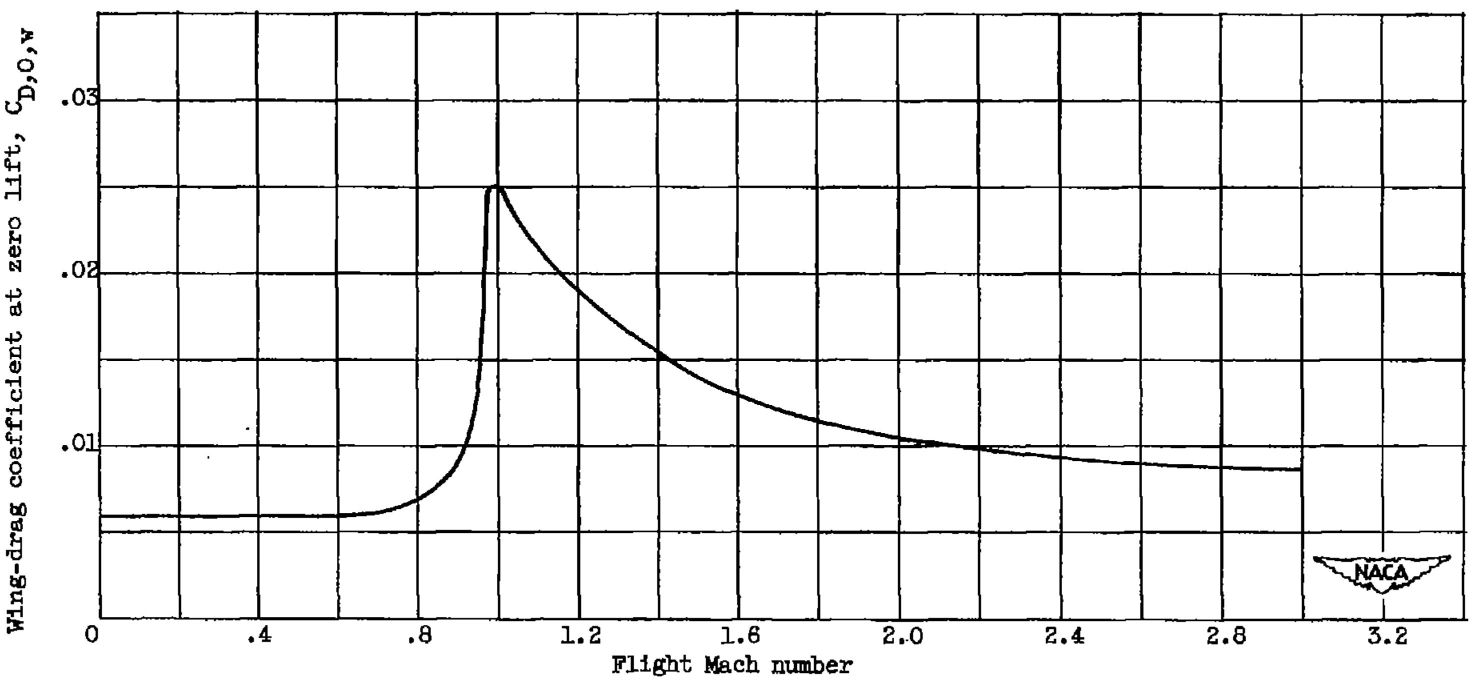

Figure 35. - Assumed variation of drag coefficient for straight wing at zero lift with flight Mach number.

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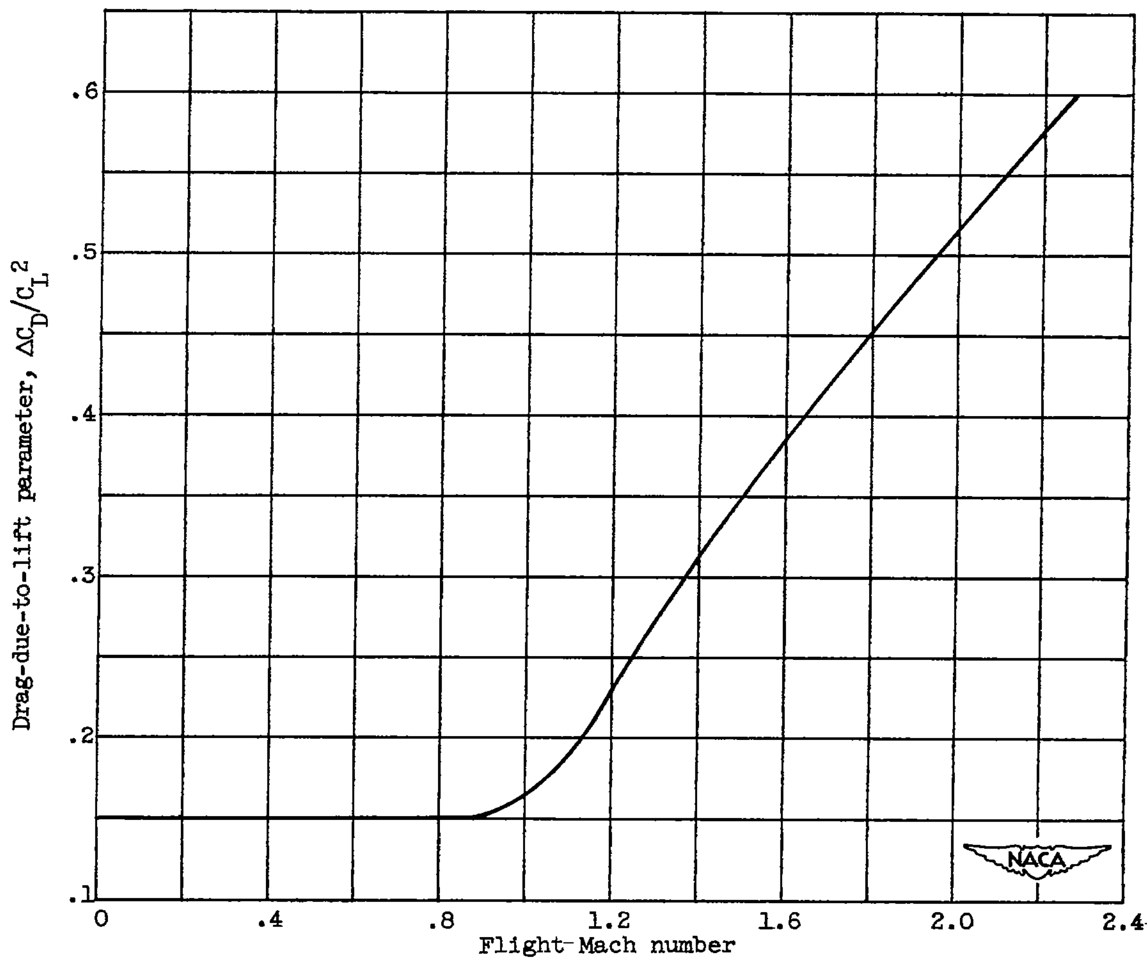

Figure 36. - Assumed variation of drag-due-to-lift parameter with flight Mach number for straight wing.

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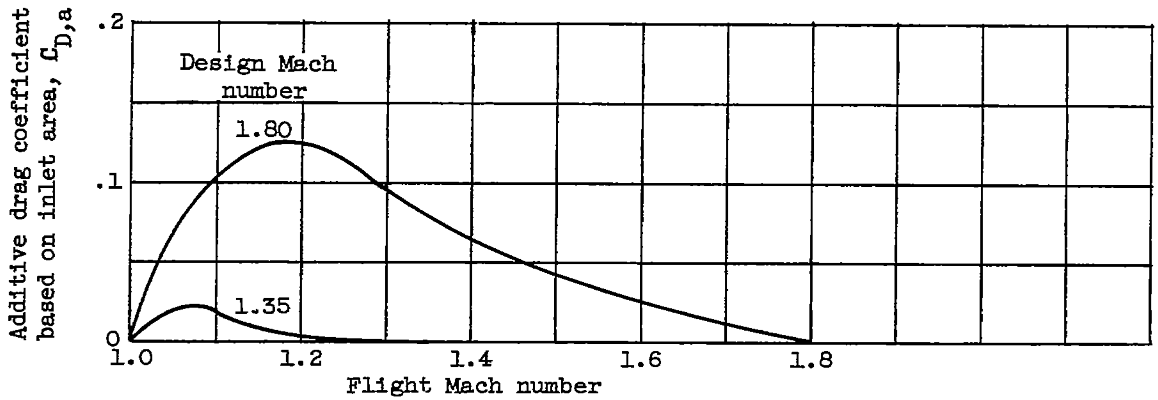

Figure 37. - Assumed additive drag coefficient as function of flight Mach number.

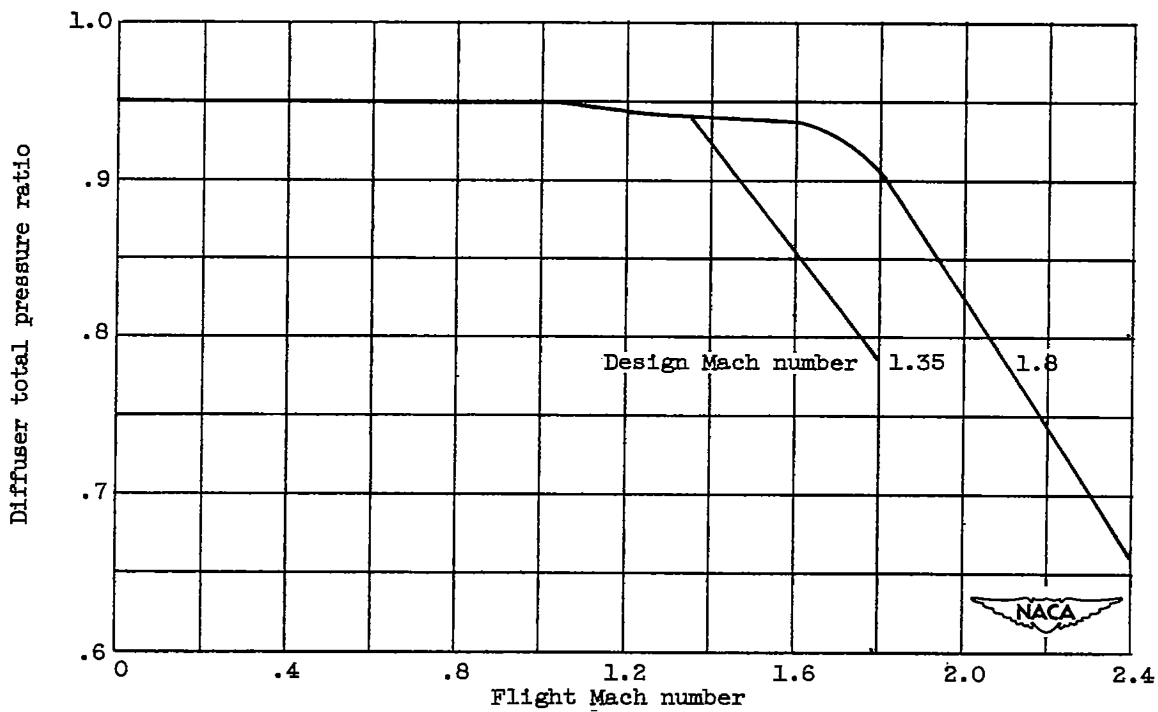

Figure 38. - Assumed inlet diffuser total-pressure recovery ratio as function of flight Mach number.

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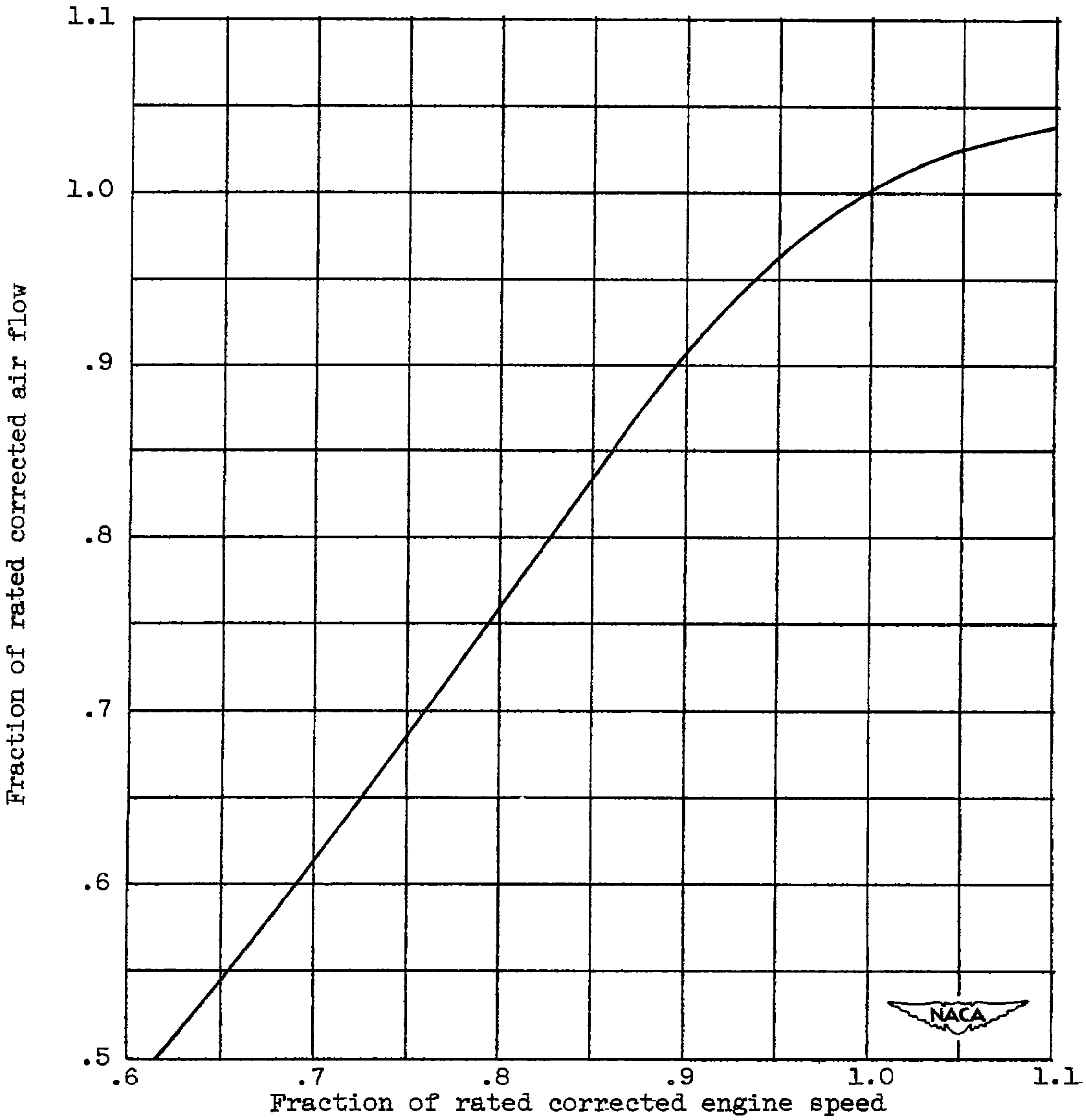

Figure 39. - Assumed variation of corrected air flow with corrected engine speed.

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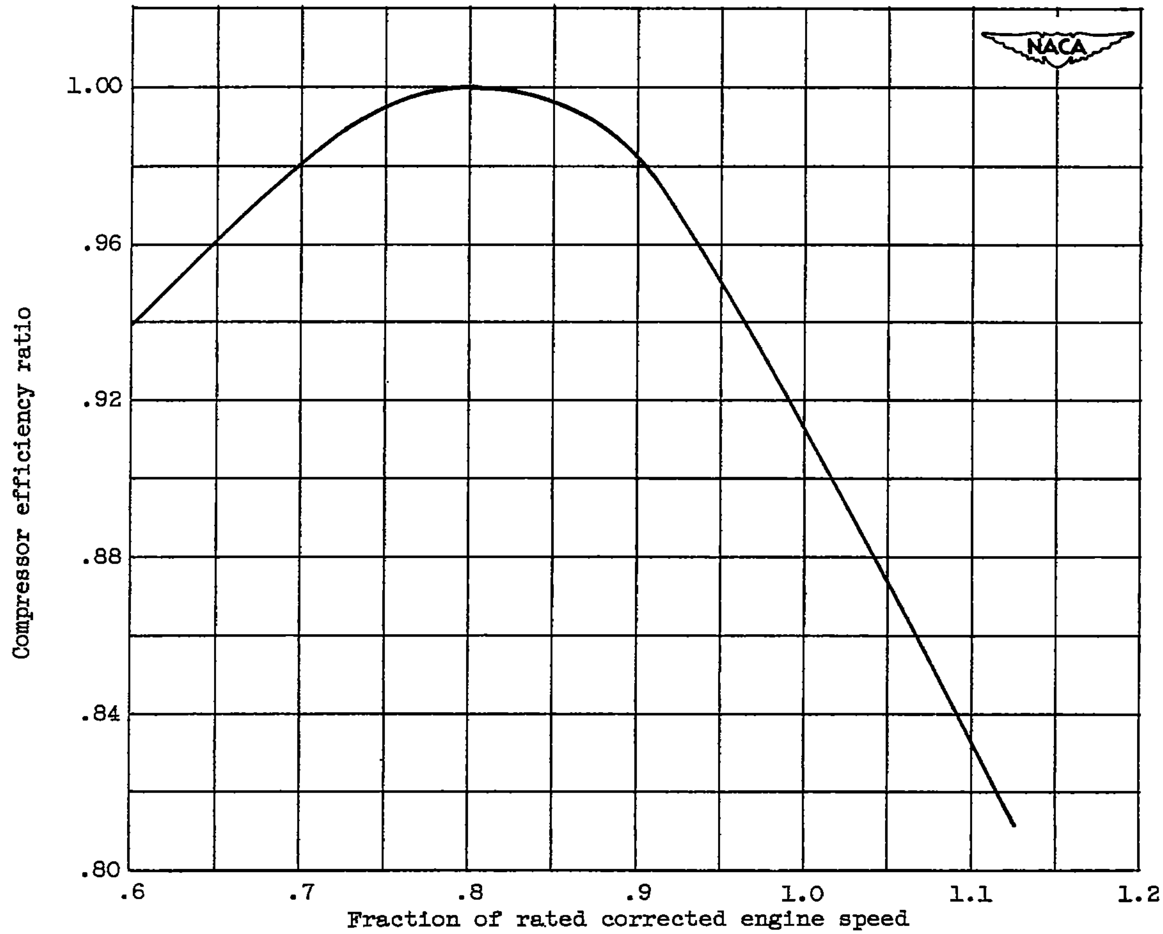

Figure 40. - Assumed variation of compressor efficiency with corrected engine speed.

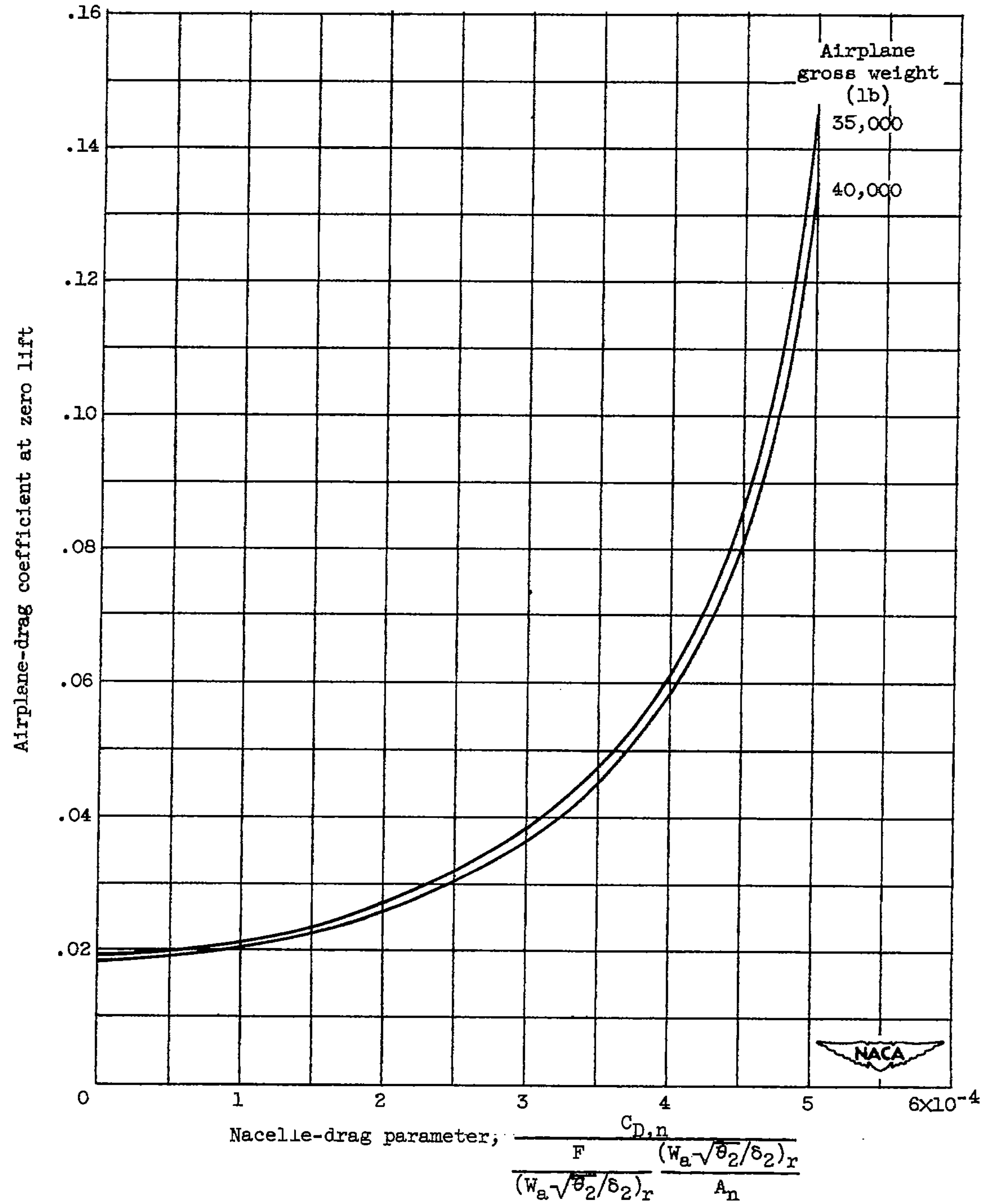

Figure 41. - Drag coefficient of short-range interceptor at zero lift as function of nacelle-drag parameter. Altitude, 50,000 feet; flight Mach number, 1.8.

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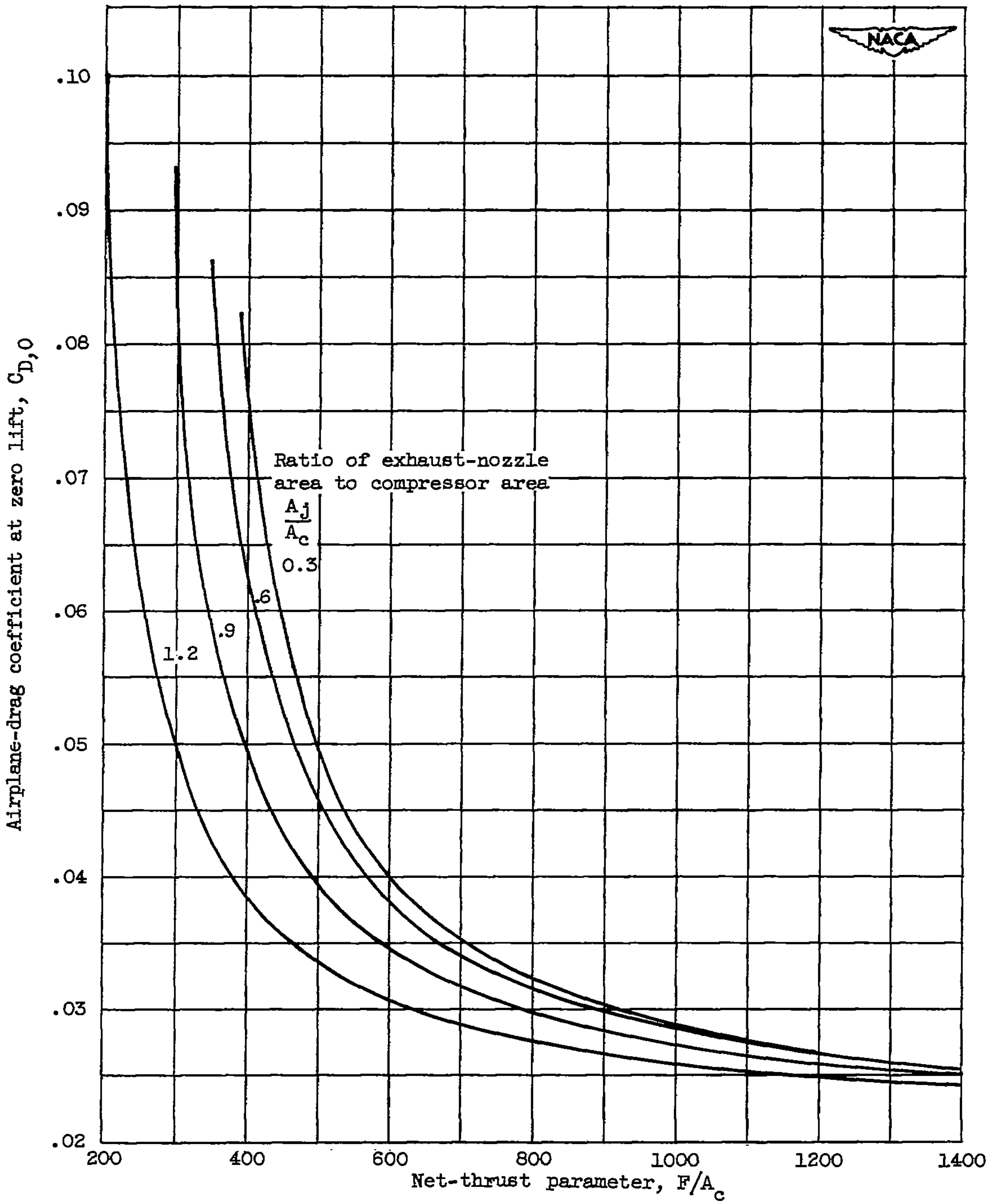

Figure 42. - Drag coefficient of short-range interceptor at zero lift (submerged engine installation) as function of engine thrust per unit compressor area. Altitude, 50,000 feet; flight Mach number, 1.8; air flow per unit compressor area, 27 pounds per second per square foot.

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